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The Rotary Wing Icing Symposium hosted by the US Army Aviation Engineering Flight Activity brought together leading experts in the field of helicopter icing from several countries. In attendance were over 130 military, government and civilian manufacturing personnel representing organizations from the United States, Canada, England, France and Germany. Presentations were given in the fields of flight testing, icing protection systems, flight operations in icing, and icing test → over

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20. Abstract

facilities. In his keynote address Paul Yaggy, Director of the Unites States Army Air Mobility Research and Development Laboratory, cited the growing emphasis for all weather operational availability. Colonel Dean E. Wright, Commander of the United States Army Aviation Engineering Flight Activity introduced the Army's Helicopter Icing Simulation System and five experimental test pilots of the Activity presented results of icing tests on the AH-1G, AH-1Q, UH-1H and CH-47C helicopters. Military requirements for helicopters capable of operating in icing conditions were discussed in a session chaired by Colonel William E. Crouch, of The Department of the Army. Colonel Horace B. Beasley of the Army Materiel Command was the moderator of discussions concerning new ice protection systems. An international flair was provided during two sessions chaired by Royal Navy Captain J. T. Checketts, British Ministry of Defence and Mr. Alan Wilson, OBE, of the Aeroplane and Armament Experimental Establishment, Boscombe Down. Icing problem areas were found to be similar among the varied types of helicopters from the different countries. Problem areas found to be common were icing of engine inlets, rotors, and windshields. Many varied approaches to solution of these problem areas were exchanged among the attendees which made the symposium a success. This summary report is prepared in three volumes. Volume I includes the symposium opening remarks, papers presented, and discussion in Session I. Sessions II and III papers and discussion are included in Volume II. The closing remarks, presentations, and discussion during Sessions IV and V are contained in Volume III.

PREFACE

The US Army Aviation Engineering Flight Activity acknowledges the outstanding participation of all who attended the Rotary Wing Icing Symposium. The papers presented by the participants were highly informative and of excellent quality. Their contributions played a significant part in the success of the symposium and achieved the aim of the conference to provide an exchange of information concerning operational and test results, testing methods and facilities, and protective measures.

PROLOGUE

Questions to authors and discussions were recorded on magnetic tape. Recording system and procedural inadequacies rendered certain portions inaudible. Mr. Hayden edited the tapes and attempted to paraphrase the comments to convey the sense of the conversation. Should any transcriptions inadequately describe the intended comment or response, please direct your wrath to Mr. Hayden and your written corrected texts to the US Army Aviation Engineering Flight Activity for literal post publication.

VOLUME III

Page

AN R&D APPROACH FOR THE SOLUTION
OF HELICOPTER ICING PROBLEMS

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ABSTRACT

The Eustis Directorate of the US Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, is actively pursuing a research and development program to establish realistic ice protection system requirements for future-generation Army helicopters and to assure that technology will be available to satisfy those requirements.

The R&D program takes into consideration the occurrence probabilities of icing conditions produced by supercooled clouds, snow, and freezing rain; the probability of Army helicopters' encountering various icing severity levels; and the penalties imposed upon the helicopter by various ice protection system concepts.

The effort includes a review of technology and conduct of trade-off analyses to allow determination of the advantages and disadvantages of various ice protection system concepts as they may apply to future-generation Army helicopters. Trade-off analyses performed will also quantify the penalties imposed by aircraft design parameters such as gross weight, payload, performance, reliability, maintainability, and costs.

The program will culminate in the flight testing of a UH-1H modified to incorporate an ice protection system representative of that required for a utility-type aircraft of the future.

Results of this R&D program are anticipated to form the basis for design and testing requirements for future-generation Army helicopter ice protection systems of maximum efficiency and minimal life-cycle costs.

INTRODUCTION

The helicopter icing problem has developed over the years along with the development of the helicopter and missions for the helicopter. Now the helicopter is on the threshold of becoming as reliable and maintainable as the fixed-wing aircraft. The performance, stability and control capabilities have been developed to the extent that future-generation Army helicopters will be capable of performing their assigned missions under all-weather conditions. All-weather capability is being required of future Army helicopters. This requirement has caused extensive soul searching to assure that the requirement can be met without adversely affecting key design goals such as reliability, maintainability, vulnerability, weight and cost effectiveness.

Although the all-weather capability requirement may require development of IFR terrain avoidance and target acquisition capabilities, the most limiting and pressing state-of-the-art improvement required is ice protection. A capability for helicopter flight in icing conditions will allow a vast majority of air mobility operations to be performed under adverse weather conditions even without IFR terrain avoidance and target acquisition capability. The most pressing need is to enable the helicopter to arrive safely and functional at its destination. Once there, mission completion is dependent upon local conditions, however, the mission can possibly be performed; but without a capability for flight in icing conditions, airmobile operations will have to be cancelled or delayed until in-route weather conditions permit. In Europe, weather statistics indicate that during winter months, airmobile operations cannot be deployed approximately 35% of the time because of icing conditions. From a strategic standpoint, this limitation is intolerable.

It therefore becomes mandatory to assure that future-generation Army helicopters possess a capability for flight in icing conditions. To meet this urgent need, to assure that accurate design criteria exist for future helicopter ice protection systems, and to assure that technology exists for meeting those criteria, the Eustis Directorate of the US Army Air Mobility Research and Development Laboratory has developed an R&D program which specifically addresses the solution of helicopter icing problems.

HELICOPTER ICING PROBLEMS

Before we begin to address the solution of helicopter icing problems, it is first necessary to define and understand the basic problem.

First and foremost, the helicopter icing problem has grown with the development of the helicopter, and it is continuing to grow as reflected in the requirements for future-generation Army helicopters. To compound the basic challenges set forth by the growing roles and missions of the helicopter, there is a serious lack of statistical data on which one can quantify the magnitude of the problem, both as it exists with current fleet helicopters and with future-generation rotary-wing aircraft.

Additionally, the forecasting procedures which have been developed over the years, based on fixed-wing aircraft icing encounter statistics, do not appear to be applicable to helicopters and especially helicopters intended for use in a wide variety of military missions. Last, but certainly not least, is the lack of worldwide meteorological data to serve as a basis for determining the statistical probabilities of encountering various icing severity levels and for assessing whether or not ice protection system design and test criteria are sufficiently accurate.

To provide a clear and thorough understanding of these basic problems, the following sections discuss each one in more detail.

Helicopter Developments

The helicopter icing problem has grown with the helicopter. When we look back over the past fifteen or twenty years, we can see, very vividly, that the helicopter has become a very useful tool--the backbone of Army air mobility. This transition did not occur overnight. Once the helicopter was introduced into the Army inventory in the 1950's, people found uses for the machine. These uses, or missions, have grown and continue to grow. During this growth process, not unlike that of the fixed-wing aircraft, we have suddenly, in the past few years, found ourselves lacking the wherewithall (technology) to satisfy the basic need. Most recent-vintage helicopters now possess the stability and control necessary to allow instrument flight (IFR). IFR capability is a fairly recent addition to helicopter growth. This capability has allowed operators to encounter icing conditions. The constant expansion of helicopter

roles and missions, has in the past resulted in some work to determine the limitations of some helicopters and their release for flight in icing severity levels termed "light to moderate icing." The CH-47 and the UH-1 have such releases.

The fact that helicopter roles and missions have grown and that the helicopter has been developed to the extent that IFR flight is now possible has forced operational commanders to repeatedly express the need for a capability to fly under forecast icing conditions. In response to this expressed need, several of the airframe manufacturers have, in the past, proposed product improvement programs to incorporate ice protection systems. In addition, in recognition of these operational needs, research and development programs have been formulated to provide technology for helicopter ice protection. These programs, although well intended, remained unfunded until very recently because of the everlasting, higher priority efforts. With the cessation of US involvement in Southeast Asia, which of course was a source of higher priority efforts, and with the requirements for future-generation Army aircraft looming in the background, emphasis is now being placed on solving the helicopter icing problem.

Future Helicopter Requirements

Very basically, since we now know that developments over recent years have demonstrated that helicopter IFR capabilities are closely approaching that of the fixed-wing aircraft, future-generation Army aircraft, such as the Utility Tactical Transport (UTTAS), Advanced Attack Helicopter (AAH), Heavy Lift Helicopter (HLH), Advanced Scout Helicopter (ASH), Light Tactical Transport (LTTAS), and others either are, or will be required to possess, an all-weather capability. The all-weather requirement is in addition to the basic mission requirements peculiar to the Army mission which involve operations within the nap-of-the-earth. This places the Army helicopter within the altitude range where icing conditions are most prevalent, increasing the frequency and probability of encountering icing conditions.

There are, of course, other mission segments such as ferry, training, and utility missions which take the Army helicopter to its service ceiling, and these missions must be considered. In some mission requirements, the service ceiling is 20,000 feet. Of primary significance is the fact that our future helicopters will also be required to be highly reliable, maintainable, invulnerable, survivable, agile, maneuverable and, most of all,

cost effective. The addition of the all-weather requirement and specifically that of ice protection can very easily have an adverse impact upon these key design goals. Our research and development objectives are, therefore, structured to assure that these factors are constantly being considered during the development of design criteria so as to optimize the criteria for minimum trade-off of these goals. The specific methodology being used under our research and development program to take these factors into consideration will be described later.

Lack of Statistical Data

Since, with few exceptions, the Army first-line helicopters do not have a capability for flying in icing conditions, statistical data is very limited. Therefore, the magnitude of the icing problem cannot, at this time, be accurately quantified. Icing problems have been discussed with several operational units in Germany. Also, the US Army Agency for Aviation Safety (USAAAVS), Fort Rucker, Alabama, has recently issued a questionnaire soliciting information pertaining to icing encounters. This is described in Reference 1 and summarized in Figure 1.

The conclusions which have been drawn to date from the limited discussions and review of the USAAAVS data are summarized as follows:

Icing Encounters Limited

Helicopter pilots, with few exceptions, have exercised sound judgement in the past by not deploying into known icing conditions. Results of the USAAAVS questionnaire to date have revealed 198 icing encounters by first-line helicopters, 170 by the UH-1, 14 by the CH-47, 13 by the OH-6 and OH-58, and 1 by the AH-1. These were, for the most part, unintentional encounters, and in most instances evasive action to egress the icing condition was initiated as soon as the encounter became evident. Immediate evasive action is not always possible, however, because of the time required to obtain air traffic control (ATC) approval of a revised IFR plan.

Reports Inconsistent

It must be emphasized here that many icing encounters have not been reported and that many of the reports are inconsistent or could be misleading. The reported icing severity levels such as trace, light, moderate, heavy, or severe can be made only qualitatively by the pilot. Since he has no real basis for quan-

tifying his assessment of severity, he has to relate the situation as he sees it. Pilot emotion is frequently included. Generally with the UH-1H, when the windshield is obscured, the severity level is reported as moderate or severe. If the windshield wiper blades, door handles, and skids begin to accrete ice, the reports generally range from trace to moderate. In either of these cases, the main rotor blade may not be accumulating ice, but if the pilot cannot see to land or has to make a side-slip approach to gain visibility, he considers it a serious or severe condition. Also generally, if torque increases or vibrations are observed by the pilot, he usually reports this as a moderate or severe icing condition. The situation is severe, however, this does not imply that the icing severity level falls into that category. The basic subject of icing severity level will be discussed in more detail later.

Forecasting Procedures

The primary cause of icing encounters, which exceed the capability of the helicopter, is the accuracy of the forecast. Approximately one-fourth (22%) of the icing encounters reported to USAAAVS were under VFR flight plans. In most of these reports, freezing levels were forecast, but icing was not. On many occasions, icing conditions are forecast for specific altitudes. The IFR flight plan is assigned a presumably safe altitude; however, icing is frequently encountered at the assigned altitude. In many instances (9.6%), icing is encountered during climb or descent through a cloud layer. Encounters under forecast icing conditions during climb and descent constitute a majority of intentional encounters. USAAAVS data indicates that the majority of icing encounters (86%) occur during cruise flight at altitudes between 1500 and 6000 feet (76%). Under the assumption that pilots rarely deploy into forecast icing and even less frequently deploy into forecast icing greater than trace or light, the USAAAVS data implies that forecasting accuracy is less than adequate for helicopters. Personal discussions with operational units in Germany also support this conclusion.

Lack of Meteorological Data

To allow the design or development of ice protection systems for helicopters (and for that matter, fixed-wing aircraft) so that the system does not adversely impact the key design goals discussed earlier, it becomes necessary to understand the probabilities of encountering various icing severity levels. Army

Regulation (AR) 70-38 (Reference 2) requires that military equipment be designed to withstand the 99th percentile of exceedance probability for the most severe environmental condition that can exist, in the most severe region of the world, during the most severe month of the year. This requirement, in essence, recognizes that one should not attempt to design for the most severe condition that could exist and allows deletion of the last one percentile of exceedance probability. AR 70-38 also acknowledges that sufficient climatic data may not exist on which to determine the 99th percentile of exceedance probability for icing conditions.

Heretofore, aircraft ice protection systems have been designed in general accordance with criteria of Federal Air Regulation (FAR) 25 (Reference 3) and MIL-E-38453 (Reference 4).

Hardware Problems

From a Research and Development (R&D) standpoint, we must consider the basic problems that exist as outlined heretofore. Not included in previous discussions are the hardware problems that might be encountered in providing a capability for helicopter icing flight. These discussions have addressed why the problem exists, how it might influence the operational capability of future-generation helicopters if the problem is not resolved in the near future, and some of the hurdles that must be overcome before we can be sure that the problem has been overcome.

Much testing has been accomplished over the past eighteen years to determine the behavior of the helicopter in the icing environment, to identify problem areas, and to resolve these problems. This work was pioneered by the Cold Temperature Laboratory of the Canadian National Research Council, Ottawa, Ontario, Canada. In the past five years, as the need for helicopters capable of flight under icing conditions has received more visibility, the tempo of testing has steadily increased. The United Kingdom, Canada, US Air Force, US Navy, and US Army Aviation Systems Test Activity have very actively pursued identification and resolution of problems. Results of the most recent tests are to be reported during this conference; however, based on results available to date, the following conclusions can be drawn:

1. Simulated and natural icing tests performed to date supplement existing limited operational icing encounter statistics.

2. Current-inventory Army helicopters cannot safely perform missions under icing conditions. Only brief exposure can be endured without risk of damage to the aircraft and/or degradation of performance and control.

3. Future-generation helicopters must have ice protection systems to protect critical components to possess an all-weather capability. These include main rotor blade, tail rotor blades or other antitorque devices, windscreens, engine inlets and particle separators, aerodynamic surfaces, leading edges, antennas, exposed control system components, and weapon systems.

4. Main rotor blade ice protection systems are necessary for sustained operation in forecast icing conditions for several reasons:

a. To preclude damage to rotor and airframe components from lethal ice chunks being self-shed from main and tail rotor blades.

b. To preclude excessive vibration induced by asymmetric shedding of ice from main rotor blades.

c. To preclude degradation of aerodynamic performance.

d. To reduce the adverse effect that accreted ice can have on blade structural dynamics under prolonged exposure.

5. To safely perform all-weather missions and specifically missions under conditions conducive to icing, future helicopters must possess a capability for measuring icing severity level in terms relatable to the characteristics of the particular helicopter.

6. Meteorological design criteria for future-generation helicopter ice protection systems must be optimized and design practices must be refined to preclude imposing excessive penalties upon basic design parameters such as weight, reliability, maintainability, vulnerability, and cost.

THE SOLUTION OF HELICOPTER ICING PROBLEMS

As can be deduced from discussions in previous sections, helicopter icing problems fall into two basic categories: those

experienced by current-inventory helicopters and those anticipated for future-generation helicopters. It becomes very important to recognize the difference between these two problems primarily because of growth in mission requirements. The most outstanding difference in mission requirements between current and future helicopters is that of all-weather capability. This requirement, imposed on future-generation helicopters, poses a new and unique challenge to helicopter design. Not only does it require that the future helicopter possess the ability for flight without external visual reference, the stability and control characteristics to allow such flight within the nap-of-the-earth, and all-weather navigation capability, but also that it possess improved reliability, maintainability, and vulnerability and that it be cost effective. These considerations are paramount while determining the ice protection system design requirements for future-generation Army helicopters.

While the primary emphasis of the R&D program of the US Army Air Mobility Research and Development Laboratory (USAAMRDL) is focused upon the future-generation Army helicopter, the problems that the Army first-line helicopters are experiencing cannot be ignored.

THE USAAMRDL R&D EFFORT

On 24 October 1969, the Eustis Directorate of USAAMRDL obtained approval of an R&D project entitled "Operational Effects on Aircraft." The purpose of this project was to identify and resolve operational problems encountered by Army aircraft. Emphasis was to be placed on those operational problems that may reoccur or be anticipated on future-generation helicopters. The most prevalent problem known at the time was that of helicopter icing. During fiscal years 1970 and 1971, very little funding or manpower was available because of higher priority efforts; however, Mr. Meyer B. Salomonsky of the Eustis Directorate performed the initial in-house investigations necessary to establish, in April 1972, the initial Army R&D effort in the field of helicopter icing.

This initial effort was conducted under competitive contract by the Lockheed-California Company. The final report on this effort (Reference 5) was published in August 1973. During this period, using results from this initial R&D effort and further in-house studies, and as more manpower and funding were made

available, the Eustis Directorate formulated an R&D Approach for the Solution of Helicopter Icing Problems. A second contract with the Lockheed-California Company was awarded in June 1973 through competitive procurement procedures.

The Eustis Directorate approach is currently in process through a cooperative effort, funded and managed by the Eustis Directorate, combining the technical expertise and facilities of the Lockheed-California Company, the US Army Aviation Engineering Flight Activity (USAAEFA, formally USAASTA), and the Canadian National Research Council, and utilizing guidance from the US Army Training and Doctrine Command (USATRADOC).

R&D Objectives

The objectives of the Eustis Directorate R&D program are two-fold: primarily, to establish accurate ice protection system design and test criteria for future-generation Army rotary-wing aircraft and to assure that technology will be available to satisfy those requirements; and secondarily, to perform the R&D in a manner that will allow maximum benefit to the current fleet, possibly in the form of technology developments and design refinements which can be immediately applied to product-improve current-inventory helicopters.

Eustis Directorate R&D Approach

The basic approach being taken by the Eustis Directorate is summarized in Figure 2. First, through analysis of meteorological data, the probabilities of occurrence of various icing severity levels caused by flight in supercooled clouds, freezing rain and drizzle, and snow are determined. Then the ability of technology to cope with various icing severity levels in view of mission profiles and projected characteristics of future-generation Army helicopters is assessed. Mission profiles and projected characteristics of UTTAS, AAH, HLH, LTTAS, ASH, AARS, and VHLH are being used in this analysis.

Since the initial Lockheed effort concluded that, basically, technology is available to support the ice protection needs of helicopters with the exception of the main rotor blade, the ice protection needs of the main rotor system are given major emphasis in this program. However, in view of the fact that during the initial effort, the impact of freezing rain, freezing drizzle, and snow was not considered, the ice protection needs and technology for protection of other critical components must be reassessed. These components include: engines,

engine inlets and particle separators, antitorque systems, aerodynamic surfaces, leading edges, antennas, exposed control system components, and weapon systems.

Assessment of the various ice protection concepts that might be applicable to main rotor blades of conventional and advanced, metal and composite blade construction is included. This allows selection of the concept which shows the most promise of effectively satisfying the needs of future-generation Army aircraft and performance of design and test development of that concept.

The key element of the Eustis Directorate approach is the conduct of trade-off analyses outlined in Figure 3. To allow determination of the optimum design criteria for various aircraft types, it becomes necessary to consider the various icing severity levels that might be used as a basis of design; the capability of technology, present and future, to support those requirements; and the mission and type of the aircraft. During the trade-off analyses, the impact of the ice protection system itself upon such key design parameters as weight, reliability, maintainability, vulnerability, and cost must be considered. Results of the trade-off analyses will yield information for use by procurement agencies in stipulating ice protection design criteria that are both obtainable and cost effective. Also, information produced by the trade-off analyses can be used in the future by helicopter manufacturers to assist in their selection of the most feasible and cost-effective concepts for various applications. Of course, the primary use of this information will be for selection of the recommended design criteria and selection of the most promising rotor blade ice protection concept for development and testing under this program.

<u>Flight Tests</u>

Upon completion of analyses and testing to determine the most feasible ice protection system concept and to assure that the concept can be implemented, theoretical analyses and system design parameters selected must be evaluated in a very thorough manner. To do this, a UH-1H helicopter will be equipped with an ice protection system considered to be representative of that required for application to utility-type aircraft of the future. The UH-1H is selected for this role because of its prevalence in the Army inventory and the high probability for direct application of the ice protection systems developed, to product-improve the UH-1 fleet, if deemed appropriate by cognizant program managers.

It is envisioned that the flight test program will be conducted in four phases: airworthiness demonstrations, simulated icing tests, natural icing tests, and operational evaluations.

The modified test aircraft will have to possess essentially the same flight envelope as the basic UH-1H. Minor exceptions may include empty weight, range, or payload because of the experimental nature of the program requiring flight test instrumentation and implementation of modifications at minimum cost. The minor exceptions are not expected to adversely impact program results. The envelope will be expanded and performance changes will be determined during airworthiness testing.

When the test aircraft has been found airworthy, simulated icing tests in various helicopter ground operating and flight modes will be conducted using the icing simulation facilities of the Canadian National Research Council and USAAEFA.

To assure correlation between simulated and natural icing conditions and to obtain a final verification of the capabilities and limitations of the ice protection concept developed, it is necessary to expose the test aircraft to natural icing conditions. During such tests, it is essential that the severity of the icing condition be known. This requires a capability for onboard measurement of various parameters such as ambient temperature, liquid water content, and, if possible, water droplet size--or as a minimum, ice accretion rate.

After completion of the flight test program, the design criteria established through analytical means will be reassessed to assure that optimum design criteria can be recommended.

Operational Evaluation

After the design criteria have been verified or adjustments made as a result of the engineering flight test program, the Eustis Directorate R&D approach includes the conduct of an operational evaluation. This would entail the modification of several (ideally eight to ten) operational aircraft to incorporate ice protection systems, and the deployment of these aircraft to an operational unit within the icing belt for their use. The operational evaluation would be closely monitored by an engineering team to allow accurate determination of design inadequacies, reliability and maintainability limitations, or other problems. Information from this operational evaluation will

be used to update, as necessary, design or test criteria previously established, and to identify any other problems that require resolution prior to implementation on production, future-generation helicopters.

Results to Date

Although it is not the intent of this paper to discuss results of the Eustis Directorate R&D program, since the final analysis and end result will not be known until mid-1975, I will outline some of the basic accomplishments to date.

Design Criteria Established

The Lockheed-California Company has completed the basic elements of the program by determining the recommended design criteria for future-generation Army aircraft and for the test UH-1H helicopter. The findings are presented in Figure 4. These findings are based on meteorological data obtained from the National Weather Service Records Center, Ashville, North Carolina. Of significance is the fact that the most severe meteorological design criteria for an exceedance probability of 0.01 are less demanding than the requirements of FAR 25 (Reference 3) or MIL-E-38453 (Reference 4) at mean effective droplet sizes below 27 microns, but they are slightly more demanding for larger droplet sizes. The important point is the fact that criteria differ. Also these criteria cover supercooled clouds and not freezing rain or drizzle. Criteria for freezing rain and drizzle have been developed; however, it has been concluded, through trade-off analyses, that to meet these criteria will not be feasible in most applications. The capability and limitations of the test UH-1H aircraft to cope with freezing rain and drizzle conditions will be evaluated during the flight test program.

Trade-off Analyses Completed

Lockheed has completed assessments of technology, has made trade-off analyses, and has selected the cyclic-electrothermal rotor blade ice protection system concept as the most feasible for application to most future-generation helicopters. Mission profiles and projected characteristics of future-generation helicopters were prepared by USATRADOC. One example of the type of information to be made available in final reports is presented in Figure 5. Plots shown in this figure illustrate

the weight penalties imposed by various types of future-generation aircraft ice protection systems as a function of design mission duration. Information of this type, combined with other parametric trade-off analyses, allows ready determination of the type of ice protection system most suitable for the specific need.

Test Aircraft Design

Lockheed has completed the preliminary design for incorporation of ice protection systems on the UH-1H test aircraft and is in the process of finalizing the detail design for incorporation of modifications on the test aircraft. Reference 6 discusses this subject in depth.

Engineering Flight Test Plans

Test plans have been prepared jointly by Lockheed and USAAEFA, and testing under a combined icing test program is currently scheduled to begin about 6 January 1975 in Ottawa, Canada. Test plans are directed toward achieving, during the 1974-75 winter testing season, ten productive flight test hours of simulated icing tests using the Ottawa Spray Rig and the USAAEFA Helicopter Icing Spray System and approximately fifteen productive flight test hours in natural icing conditions.

Operational Evaluations

Planning for operational evaluations to be conducted in Germany during the winter of 1975-76 is in progress; however, conduct of these tests to obtain meaningful results is dependent upon adequate funding. In addition, these plans depend to a large extent on the success of currently planned simulated and natural icing engineering flight tests.

Shortcomings

Two shortcomings have become apparent during work performed to date that are not totally or directly relatable to the Eustis Directorate R&D Program. They are, however, relatable to the ultimate solution to the helicopter icing problem and therefore are discussed herein.

Forecasting Procedures

As mentioned earlier, the primary cause of icing encounters which exceed the capability of the helicopter, is the accuracy

of the forecast. While it is not the intent of the Eustis Directorate R&D program to revise forecasting procedures, it is essential for us to understand these procedures so that design requirements to be stated for future helicopter ice protection systems will be compatible with the real world, operational environment.

The Army relies upon the US Air Force Air Weather Service for weather forecasts. The Air Weather Service utilizes every source of data available to provide constantly updated weather forecasts. These sources include the US Weather Bureau and participating weather services of other nations, US military and commercial weather stations located about the free world, and weather satellites. Data from all these sources is fed into the Air Weather Service central facility in Fort Worth, Texas, where very elaborate computer facilities are located for processing the vast amount of data. An elaborate network of tie lines exist to feed processed data to each of the Air Weather Service Stations located essentially worldwide. This includes the Global Weather Center in Omaha, Nebraska.

Forecasters within the individual weather stations receive an overview of the weather trends, updated every six hours and more frequently when required, for their area of responsibility. The forecaster evaluates the information, injects his own knowledge and experience of local trends, and generates a forecast.

Specifically for icing, the Air Weather Service has developed a very methodical forecasting means. Icing forecasts are based on many factors, but primarily on ambient temperature and dew-point spread. Figure 6 presents a flowchart which outlines, step by step, the factors that the forecaster considers to establish an icing forecast. As can be seen from this chart, the forecaster first considers the type of condition prevailing, i.e., whether or not clouds exist and the type of cloud (nonfrontal, warm front, or cold front). Then he determines whether or not precipitation is occurring or is anticipated and the degree of precipitation. To simplify this discussion, let us examine only the nonfrontal cloud condition with no precipitation. The forecaster, for this weather condition, simply determines the temperature range (there are four listed with two dew-point spreads for each), the advection characteristics, and the cloud type, and he has a good indication of the icing condition to be forecast in terms of no icing, trace rime, light rime, light

clear, or light mixed. It is of interest to note that the only condition which will create a severe icing forecast is freezing rain. Moderate icing can be encountered only under frontal and freezing drizzle conditions.

This system of forecasting has proven to be quite adequate for fixed-wing aircraft. Why then is it not adequate for the helicopter? The answer to this question becomes very apparent once the basis for this rather simple forecasting techniques is understood. The Air Weather Service has developed this methodology through statistical analysis of pilot reports of icing conditions being encountered at a specific time and place where the meteorological conditions were known. Very simply, the Air Weather Service has correlated meteorological conditions to pilot opinion, and now they are relating the condition in terms that the pilot understands. The only real problem introduced here is that the pilot reports that were analyzed and included in the statistical analysis were from fixed-wing aircraft of various types, most of which had operable ice protection systems installed. Can these weather forecasts be used by helicopter operators? Apparently not. This conclusion is drawn because of the large number of inadvertent encounters when icing was not forecast and because of the many negative encounters when icing conditions were forecast. This conclusion is also drawn because of the realization that ice accretion rate depends on many parameters, such as airspeed, contour of various aerodynamic surfaces, porosity of the materiel, and whether or not ice accumulations adversely affect the aerodynamic performance of the vehicle. These are the factors that the pilot can sense and to which he relates icing severity level. From these standpoints, the helicopter differs drastically from the fixed-wing aircraft.

To rectify this shortcoming, it may become necessary for the Air Weather Service to adjust their forecasts so that the helicopter is specifically addressed. To do this, however, the Air Weather Service must receive accurate pilot reports.

Ice Detection and Cockpit Display

Work performed to date by Lockheed indicates a need for automatic or semiautomatic capability to allow efficient operation of the helicopter ice protection system and to allow the crew to concentrate on mission-related duties. An accurate ice detector is the heart of such a capability. It appears that

through foresight and efforts of the Canadian National Research Council, such an ice detector is now available. The test aircraft will be equipped with such a device.

The ice detector which is suitable for helicopter applications is very responsive. The electronic signals produced by the ice detector are relatable to ice accretion rate which may be relatable to cloud liquid water content. These parameters, when combined with ambient temperature (which will also be measured accurately on the test aircraft), can, in addition, be used as sources of information to the crew if the proper panel display is used.

One might question the need for such information, however, there are several advantages for displaying information of this type to the crew.

First, future-generation helicopters will be qualified for flight up to a particular icing severity level. This qualification will be relatable to known test conditions such as liquid water content and ambient temperature. The crew needs to assure that the helicopter is not exposed to icing severity levels beyond its capability.

Second, the display of icing severity level will assist the crew in finding a safer flight altitude where icing severity is minimal.

Third, the crew can use displayed icing severity level parameters as a basis of pilot reports to the Air Weather Service. As mentioned earlier, the Air Weather Service will need such reports to refine forecasting procedures.

Last, but certainly not least, there is a need for accurately measuring icing severity level parameters during engineering flight tests. This is especially important during natural icing tests.

The cockpit panel display, if properly configured and standardized, could serve all these purposes. Several panel display concepts have been developed and used over the years, but most are only indicators of the presence of ice. The display to be used on the test aircraft is the meter type illustrated in Figure 7. This gauge gives interpretations of icing severity level in terms of trace, light, moderate, and severe. It will be calibrated during the engineering flight

test program and may prove suitable for operational use; however, the optimum cockpit display required for operational use is not known at this time. Several concepts have been qualitatively examined to date, however, it seems that many varied opinions exist. These opinions vary from no need for cockpit display to a need for accurate readout of liquid water content, droplet size, and ambient temperature.

Discussions with instrumentation manufacturers reveal that they would undertake development of the required panel display if they knew the requirements. It is felt that the requirements of all helicopter operators will be the same, but at this time these requirements vary. It is necessary for operators to agree, at a very early date, on a panel display concept that will satisfy the current and future need so that work may begin.

One concept is illustrated in Figure 8 and is presented here for consideration. This is a relatively simple device and is very similar to the instrument landing system (ILS) indicator. The horizontal bar could indicate ambient temperature, and the vertical bar could indicate either liquid water content (LWC) or some parameter relatable to LWC, such as ice accretion rate. Unsafe, caution, and safe conditions could be indicated by color-coded, cross-hair zones. This would provide the pilot with ready reference to the capability of his aircraft to cope with a forecast condition or to assess if local conditions are worse or better than the forecast. It would also assist him in finding less dangerous altitudes once he has clearance to change altitude. The pilot can also feed back his findings to the Air Weather Service for update of forecasts.

CONCLUSION

Basic helicopter icing problems have been identified. It is concluded that ice protection of critical components is required to allow safe helicopter operation in the icing environment. Also, it is concluded that technology is at hand to provide ice protection of critical components with the exception of the main rotor blade.

The Eustis Directorate R&D approach is expected to resolve this wide gap in ice protection technology by first establishing realistic design criteria, gaining a thorough understanding of the penalties imposed by the ice protection system

in the form of parametric trade-offs, and developing a valuable research tool in the form of an ice-protected and thoroughly instrumented UH-1H helicopter. This helicopter can and will be used to further identify and resolve remaining helicopter icing problems both through engineering flight testing and operational evaluations.

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3. Federal Aviation Regulation, Part 25: "Airworthiness Standards," Appendix C: Transport Category Airplane.

4. MIL-E-38453 (USAF) Amendment 1, "Environmental Control, Environmental Protection, and Engine Bleed Air Systems, Aircraft and Aircraft Launched Missiles, General Specification For," 4 May 1967.

5. Werner, J. B., ICE PROTECTION INVESTIGATION FOR ADVANCED ROTARY WING AIRCRAFT, USAAMRDL Technical Report No. 73-38, US Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, August 1973.

6. Werner, J. B.; Schmidt, K. K.; ADVANCE ELECTROTHERMAL DEICING FOR AN EXPERIMENTAL ARMY HELICOPTER, Lockheed-California Company, presented to the Rotary Wing Icing Symposium, 4-6 June 1974.

ICING ENCOUNTER STATISTICS
USAAAVS - as of 12 March 1974

TOTAL REPORTS	198	UH-1 REPORTS	170
MISSION		**VIBRATION**	
TRAINING	101	SLIGHT	26
SERVICE	67	MODERATE	12
FLIGHT PLAN		SEVERE	9
IFR	136	NONE	121
VFR	33	**SEVERITY**	
ALTITUDE		TRACE	25
1500'-6000'	148	LIGHT	82
AIRSPEED		MODERATE	52
80 - 100 kt	111	HEAVY	10
TYPE ICE		**PROBLEMS**	
RIME	124	CONTROL	27
CLEAR	40	VISIBILITY	81

Figure 1

APPROACH

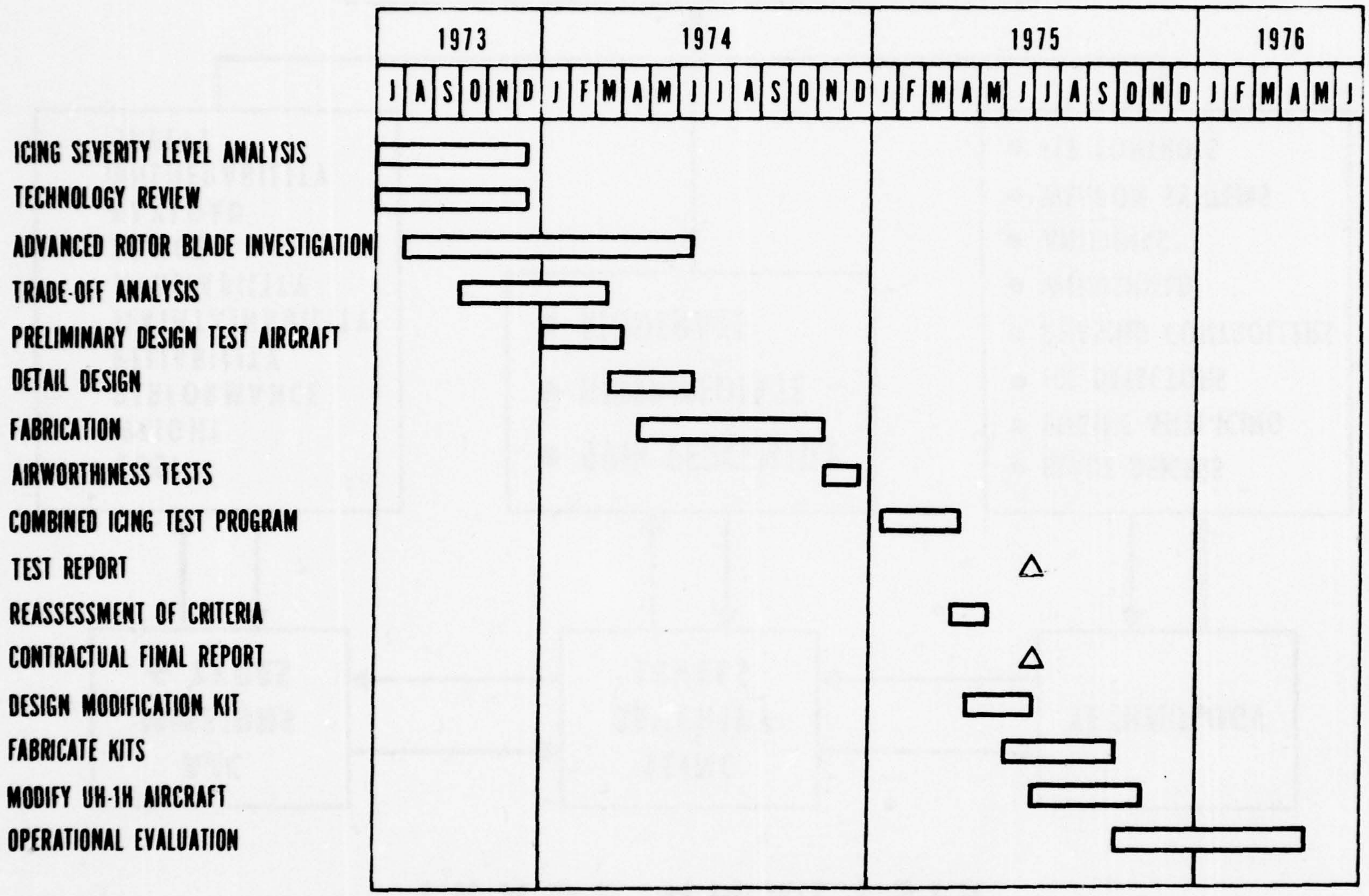


Figure 2

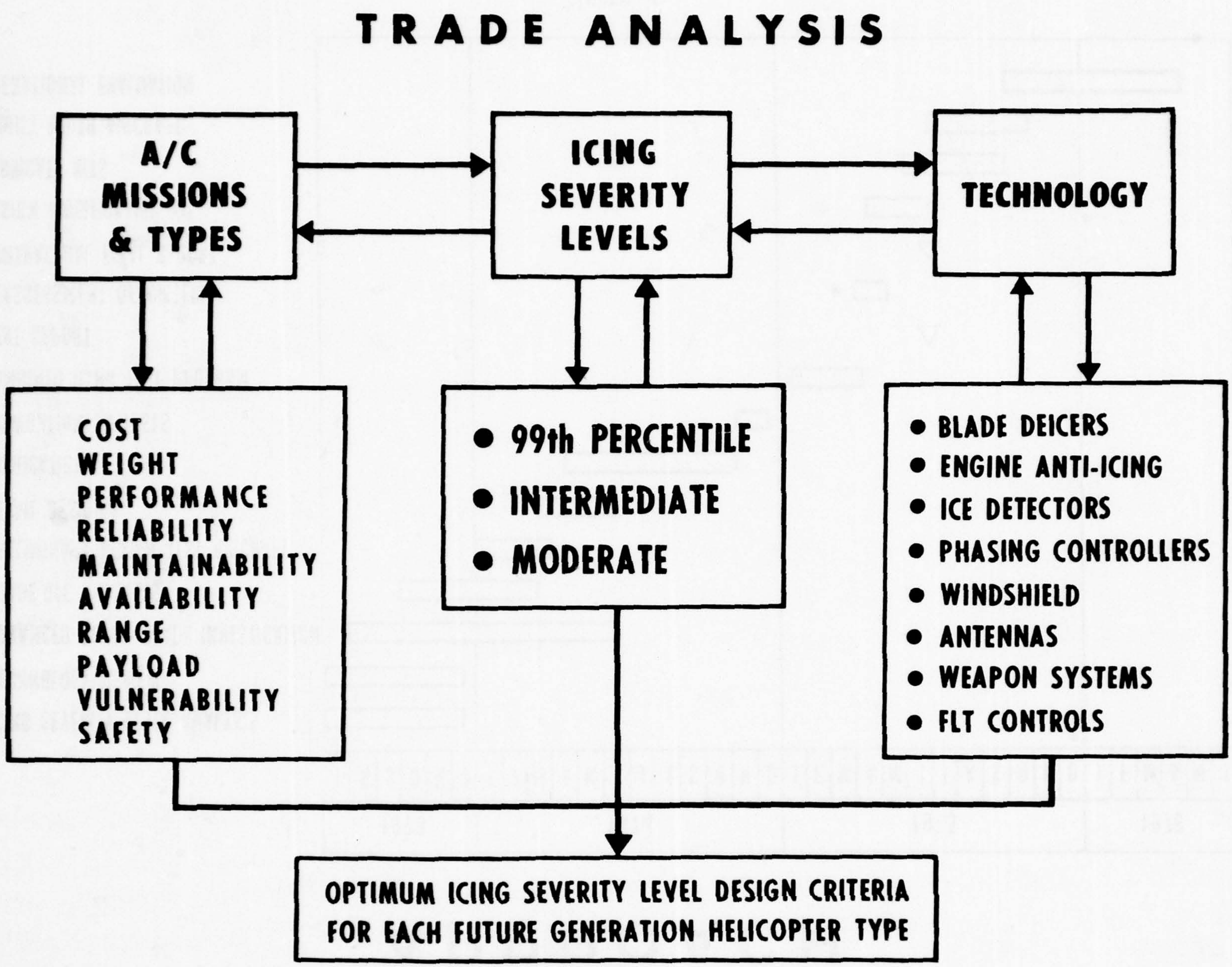

TRADE ANALYSIS
A/C MISSIONS & TYPES
ICING SEVERITY LEVELS
TECHNOLOGY
COST
WEIGHT
PERFORMANCE
RELIABILITY
MAINTAINABILITY
AVAILABILITY
RANGE
PAYLOAD
VULNERABILITY
SAFETY
99th PERCENTILE
INTERMEDIATE
MODERATE
BLADE DEICERS
ENGINE ANTI-ICING
ICE DETECTORS
PHASING CONTROLLERS
WINDSHIELD
ANTENNAS
WEAPON SYSTEMS
FLT CONTROLS
OPTIMUM ICING SEVERITY LEVEL DESIGN CRITERIA FOR EACH FUTURE GENERATION HELICOPTER TYPE

Figure 3

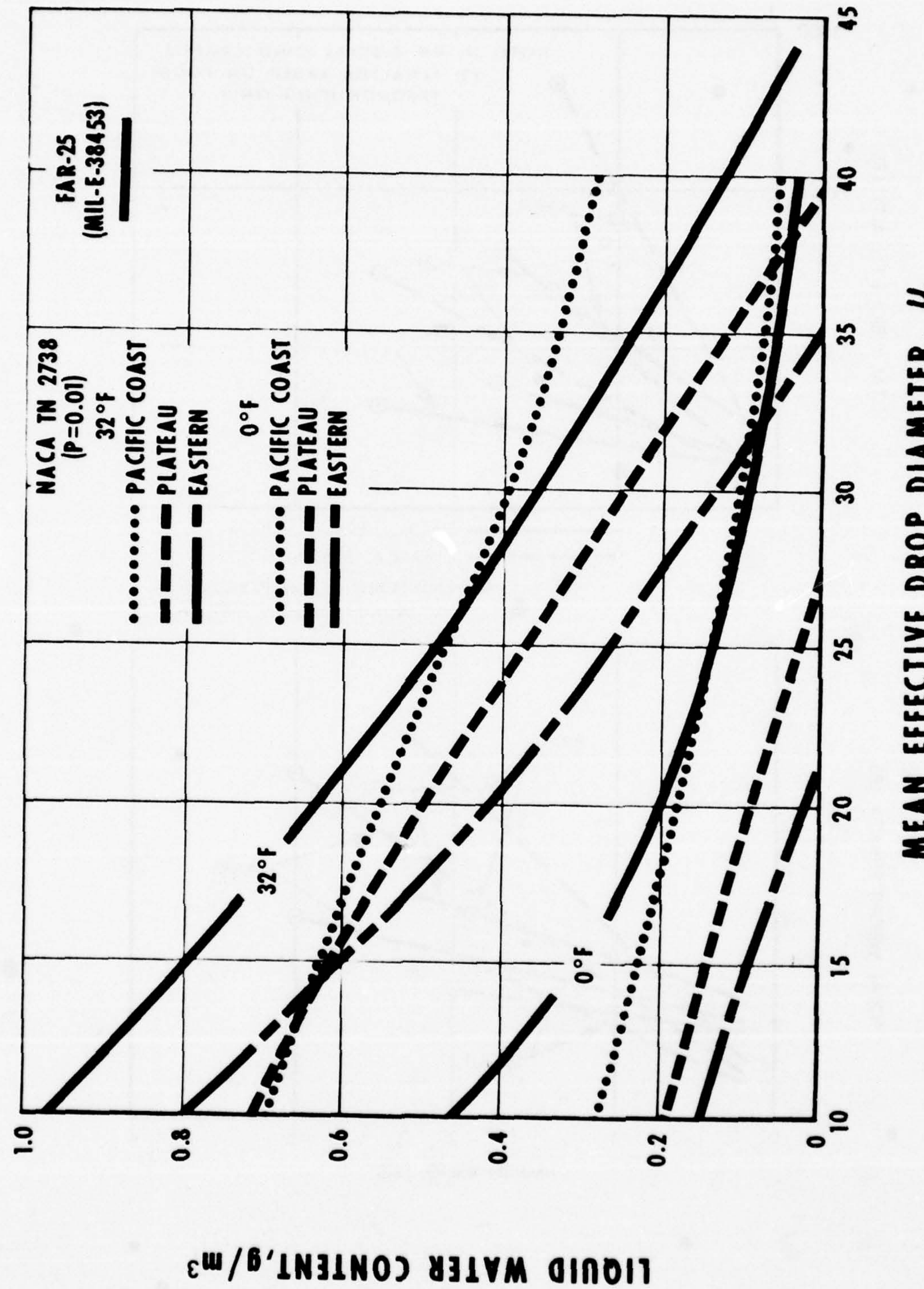

NACA TN 2738
(P=0.01)
32°F
PACIFIC COAST
PLATEAU
EASTERN
0°F
PACIFIC COAST
PLATEAU
EASTERN
FAR-25
(MIL-E-38453)
32°F
0°F
LIQUID WATER CONTENT, g/m3
MEAN EFFECTIVE DROP DIAMETER, μ
0
0.2
0.4
0.6
0.8
1.0
10
15
20
25
30
35
40
45

Figure 4

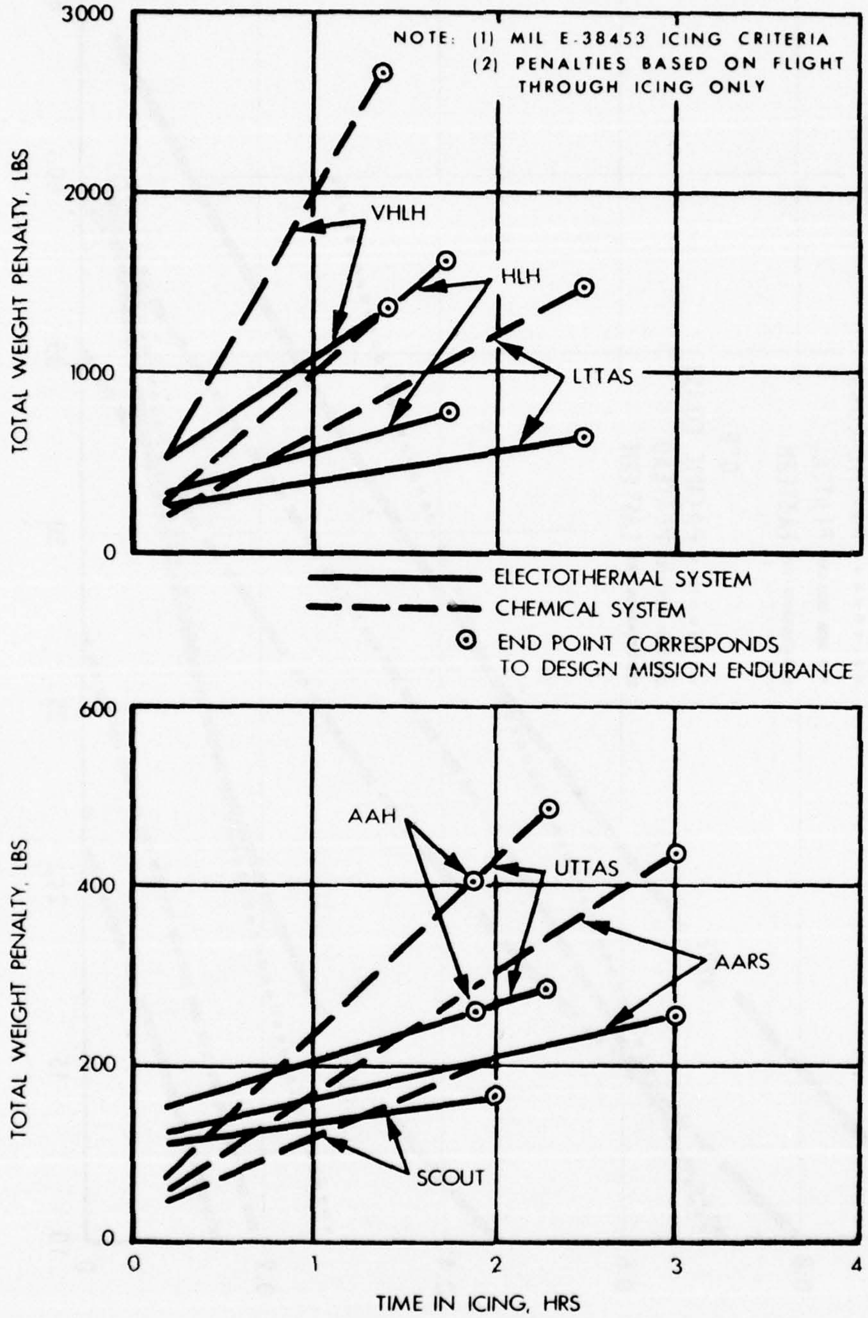

NOTE: (1) MIL E-38453 ICING CRITERIA
(2) PENALTIES BASED ON FLIGHT THROUGH ICING ONLY
TOTAL WEIGHT PENALTY, LBS
3000
2000
1000
0
VHLH
HLH
LTTAS
ELECTOTHERMAL SYSTEM
CHEMICAL SYSTEM
END POINT CORRESPONDS TO DESIGN MISSION ENDURANCE
600
400
200
AAH
UTTAS
AARS
SCOUT
0
1
2
3
4
TIME IN ICING, HRS

Figure 5

ICING RULES - USAF AIR WEATHER SERVICE

	PRECIPITATION	TEMPERATURE	SPREAD	ADVECTION	CLOUDS	FORECAST
In clouds not frontal	Widespread	0 to -7	>2			NO ICG
			≤2	Warm, neut, weak cold		
				Strong cold		TRACE RIME
		-8 to -15	>3			NO ICG
			≤3	Warm, neut, weak cold		
				Strong cold		TRACE RIME
		-16 to -22	>4			NO ICG
			≤4			TRACE RIME
		-22				NO ICG
	None	0 to -7	>2			
			≤2	Warm		
				Neut, weak cold		TRACE RIME
				Strong cold	ST	LIGHT RIME
					CU	LIGHT CLEAR
					Vig CU	
		-8 to -15	>3			NO ICG
			≤3	Warm		
				Neut or weak cold		TRACE RIME
				Strong cold	ST	LIGHT RIME
					CU	LIGHT CLEAR
					Vig CU	LIGHT MIXED
		-16 to -22	>4			NO ICG
			≤4			TRACE RIME
		-22				NO ICG
Warm front or occlusion	None	-15				LIGHT RIME
		-1 to -15			ST	
		0 to -8			CU	LIGHT CLEAR
		-9 to -15			CU	LIGHT MIXED
Cold front or deep low	None	-15			ST	MODERATE RIME
		-1 to -15			ST	
		0 to -8			CU	MODERATE CLEAR
		-9 to -15			CU	MODERATE MIXED
Freezing	Drizzle	0				MODERATE CLEAR
	Rain	0				SEVERE CLEAR
Added	None	0 to -7	<2	Neut or weak cold	Below inversion	LIGHT RIME

Figure 6

TEST AIRCRAFT DISPLAY

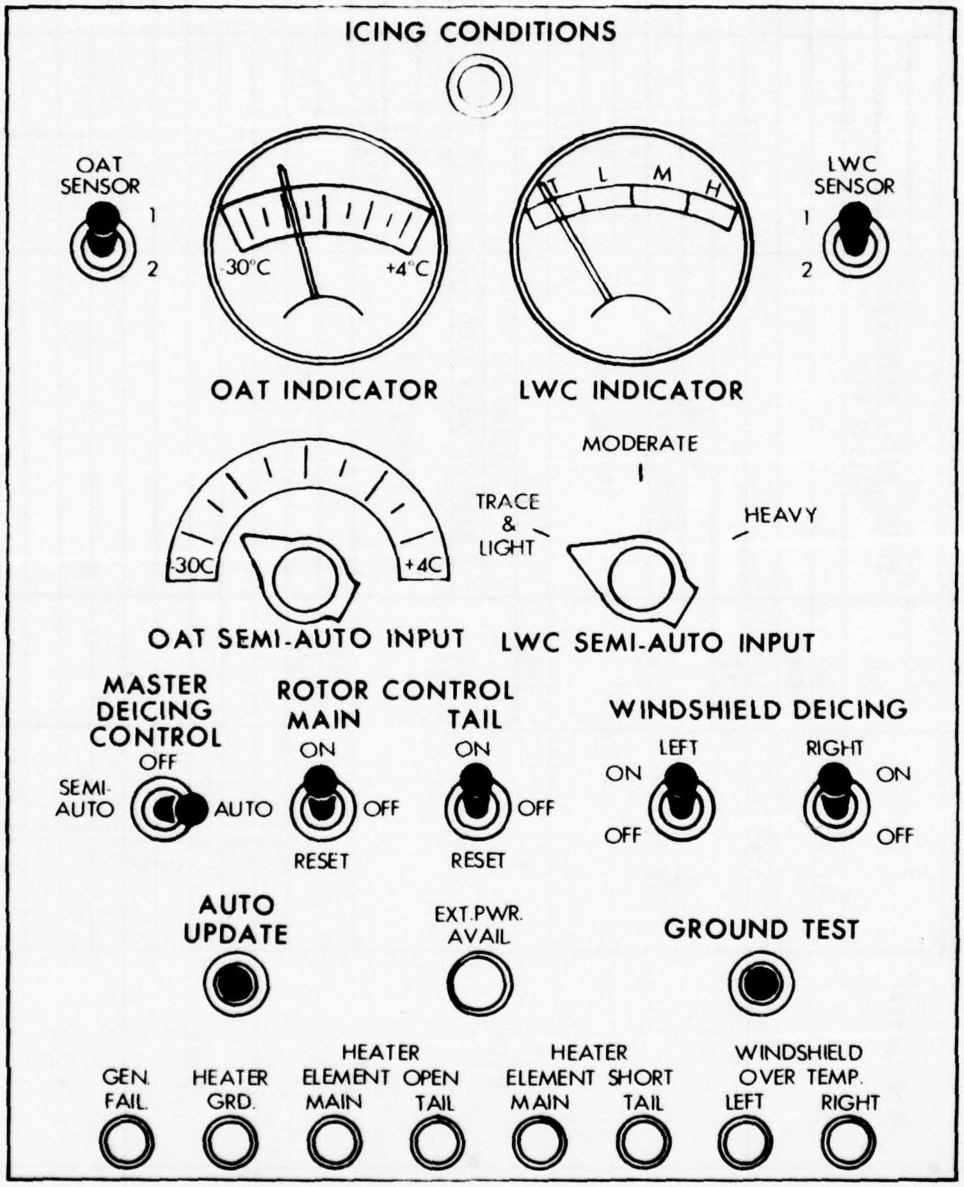
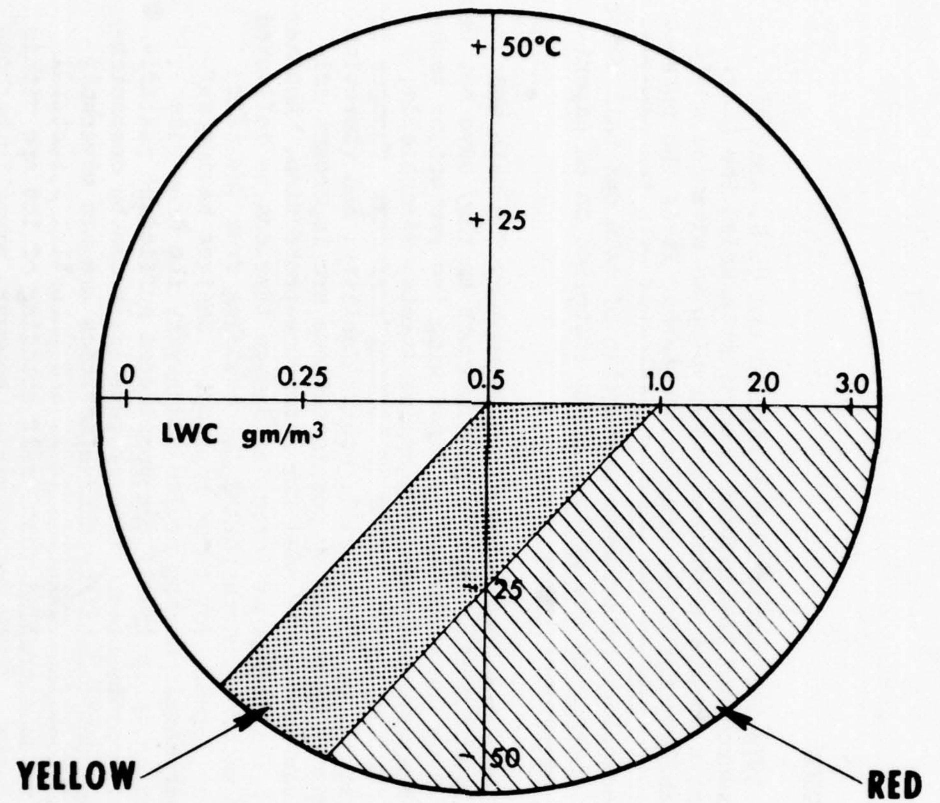


Figure 7



PROPOSED PANEL DISPLAY FOR ICING SEVERITY LEVEL

Figure 8

AN ADVANCED ELECTROTHERMAL DEICING SYSTEM FOR AN EXPERIMENTAL ARMY HELICOPTER

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Program Leader
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Capability Program

Kenneth K. Schmidt,
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Engineer

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INTRODUCTION

On June 30, 1973 the Eustis Directorate of the U. S. Army Air Mobility Research and Development Laboratory awarded the Lockheed-California Company a contract to develop an advanced anti/-deicing capability for U. S. Army helicopters. It is the purpose of this paper to present the results associated with the requirements and technology relating to protection of main and tail rotor blades and to describe the systems being installed on an experimental Army helicopter (a UH-1H).

For helicopters, the requirements and technology for main and tail rotor ice protection are unique inasmuch as they have not been developed during the evolution of fixed wing ice protection technology, as has been the case with engine inlets, windshields, flight probes, and leading edges of lifting surfaces. Changes in aircraft drag, lift, pitch, roll, controllability, and vibration levels which may be caused by ice formations are important criteria in establishing the need for rotor ice protection. Another criterion is the degree of potential damage that can be inflicted on the main rotor by uncontrolled ice shedding from the tail rotor or vice versa. For those helicopter designs having sufficient power margin during cruise and hover, the drag rise caused by icing is in itself not considered sufficient justification for ice protection. This is true if it can be demonstrated that the controllability characteristics are not adversely affected and that the vibration level and potential structural damage due to uncontrolled rotor self-shedding of ice are within tolerable limits. It is not possible, however, normally to control the degree of self-shedding which will occur. There is thus a substantial danger of asymmetric shedding, and this indeed turns out to be the greatest problem associated with not providing ice protection on the main rotor blades. The natural self-shedding that may occur is strongly related to the blade skin temperature which is a function of the ambient temperature and the distance along the rotor span. This distance determines

the degree of aerodynamic heating. As the blade temperature and hence the bond interface temperature decrease, the adhesive strength of ice to the surface increases. Thus, the lower the ambient, the heavier the buildup of ice that may occur before self-shedding will be realized. Therefore, the likelihood of severe vibrations due to asymmetric shedding increases as the ambient temperature goes down. On most, if not all helicopter models, attempts to induce symmetrical ice shedding by rapidly varying main rotor speed and by pumping the collective, and/or rapid cyclic control pulse inputs either failed or resulted in a greater vibration level. As a result, there is no significant evidence that true all weather capability can be obtained on a helicopter without rotor blade ice protection.

A review of technology has indicated that although 25 years of testing has been devoted to attempting to gain a satisfactory rotor blade icing protection concept, the goal has been elusive. Past systems suffered from either excessive weight, unsatisfactory performance characteristics, or very poor reliability and service life. One of the principal objectives of the present program, therefore, is to advance the state-of-the-art of protection and to develop an ice protection system concept which is lightweight, functionally efficient, and mechanically and electrically reliable. It is the purpose of this presentation to: (1) briefly review the candidate ice protection system concepts which have been considered, (2) present the concept which has been selected for development and indicate the rationale for that selection, (3) describe the complete aircraft system which is to be installed on the UH-1H and tested this coming winter at Ottawa, and (4) to describe the anticipated test program.

CANDIDATE ROTOR ICE PROTECTION CONCEPTS

A summary of the advantages and drawbacks of various ice protection techniques is shown in Table I. The principal feasible methods of rotor ice protection were reduced to three during the present study: (1) Chemical freezing point depressant system, (2) thermal anti-icing using engine exhaust air as the heat source, and (3) electrothermal cyclic deicing. Pneumatic rubber boots employed on fixed wing aircraft do not appear appropriate on rotor blades due to unacceptable life and severe aerodynamic effects. Icephobic tapes and films have also generally been found disappointing from the points of view of providing the required low adhesion strength and satisfactory life.

Figure 1 shows how the freezing point system could be applied to a rotor system. Commonly employed fluids are alcohol and glycol.

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Table I

ROTOR ICE PROTECTION CONCEPTS

ALTERNATIVE	MECHANISM	ADVANTAGES	DISADVANTAGES
CHEMICAL FREEZING POINT DEPRESSANT	SUITABLE FLUID SUCH AS ALCOHOL-GLYCERIN MIX IS INJECTED THROUGH HOLES, NOZZLES OR POROUS LEADING EDGE PANELS – DEPRESSING FREEZING POINT – PREVENTING OR REMOVING ICE	LOW DRY WEIGHT PENALTY LOW FAIR WEATHER DRAG SIMPLE CONTROL SYSTEM LOW COST	FIRE HAZARD OF STORED ALCOHOL BLADE IMBALANCE AND VIBRATION DUE TO FAULTY LIQUID DISTRIBUTION CAUSE DISCOMFORT AND REDUCE OPERATING LIFE HIGH VULNERABILITY TO BATTLE DAMAGE BECAUSE OF EXPOSED AREA SUBJECT TO LEAKS LOGISTICS REQUIREMENTS UNCERTAIN PROTECTION
MECHANICAL ICE REMOVAL	USE OF PNEUMATICALLY INFLATED BOOTS TO PHYSICALLY BREAK UP ICE	LOW POWER PENALTY OF ACTIVATION MODERATE WEIGHT PENALTY LOW COST ACQUISITION SIMPLE CONTROL SYSTEM	PERFORMANCE FALLS OFF WITH LOWER TEMPERATURE NO DEVELOPMENT EXPERIENCE ON HELICOPTER BLADES (UNCERTAINTY OF BOOT BOND IN CENTRIFUGAL FORCE FIELD) MODERATE FAIR WEATHER POWER PENALTY DUE TO INCREASED AIRFOIL DRAG RETRACTABILITY OF BOOT QUESTIONABLE AT ROTOR TIP DUE TO 800 "G" FIELD SEVERE EFFECT ON BLADE AERODYNAMICS WHEN INFLATED HIGH MAINTAINABILITY COST

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Table I (cont'd)

ROTOR ICE PROTECTION CONCEPTS (CONT)

ALTERNATIVE	MECHANISM	ADVANTAGES	DISADVANTAGES
AIR HEATED ANTI-ICING	HEATED AIR IS CIRCULATED THROUGH THE INTERIOR OF THE BLADE TO PREVENT ICE BUILDUP	GOOD PROTECTION LOW WEIGHT USING BLEED AIR	HIGH POWER PENALTY DUE TO BLEED AIR CONSUMPTION HIGH WEIGHT PENALTY IF COMBUSTION HEATERS ARE USED HIGH ROTOR BLADE COST DUE TO INTERNAL MANIFOLDING AND HIGH BONDING TEMPERATURES REQUIRED PROBABLE UNACCEPTABLE THERMAL STRESSES LARGE ROTATIONAL SEALS
LIQUID HEATED ANTI-ICING	ETHYLENE GLYCOL OR OIL IS HEATED IN AN ENGINE EXHAUST HEAT EXCHANGER AND CIRCULATED THROUGH TUBES INSIDE BLADE LEADING EDGE TO PREVENT ICE BUILDUP	COMPETITIVE WEIGHT PENALTY FULLY EVAPORATIVE PROTECTION	LEAKAGE INTO BLADES AND MAINTENANCE TO AVOID LEAKAGE BLADE IMBALANCE DUE TO LEAKAGE VULNERABLE TO BATTLE DAMAGE
ELECTRICALLY HEATED DE-ICING	ELECTRICALLY HEATED ELEMENTS EMBEDDED IN A PLASTIC LAMINATE AND PROTECTED BY A METAL EROSION SHIELD	GOOD PERFORMANCE MINIMUM VIBRATION BEST FLIGHT CHARACTERISTICS LOW BLADE DRAG	MODERATE POWER PENALTY MODERATE MAINTENANCE COSTS MODERATE WEIGHT PENALTY

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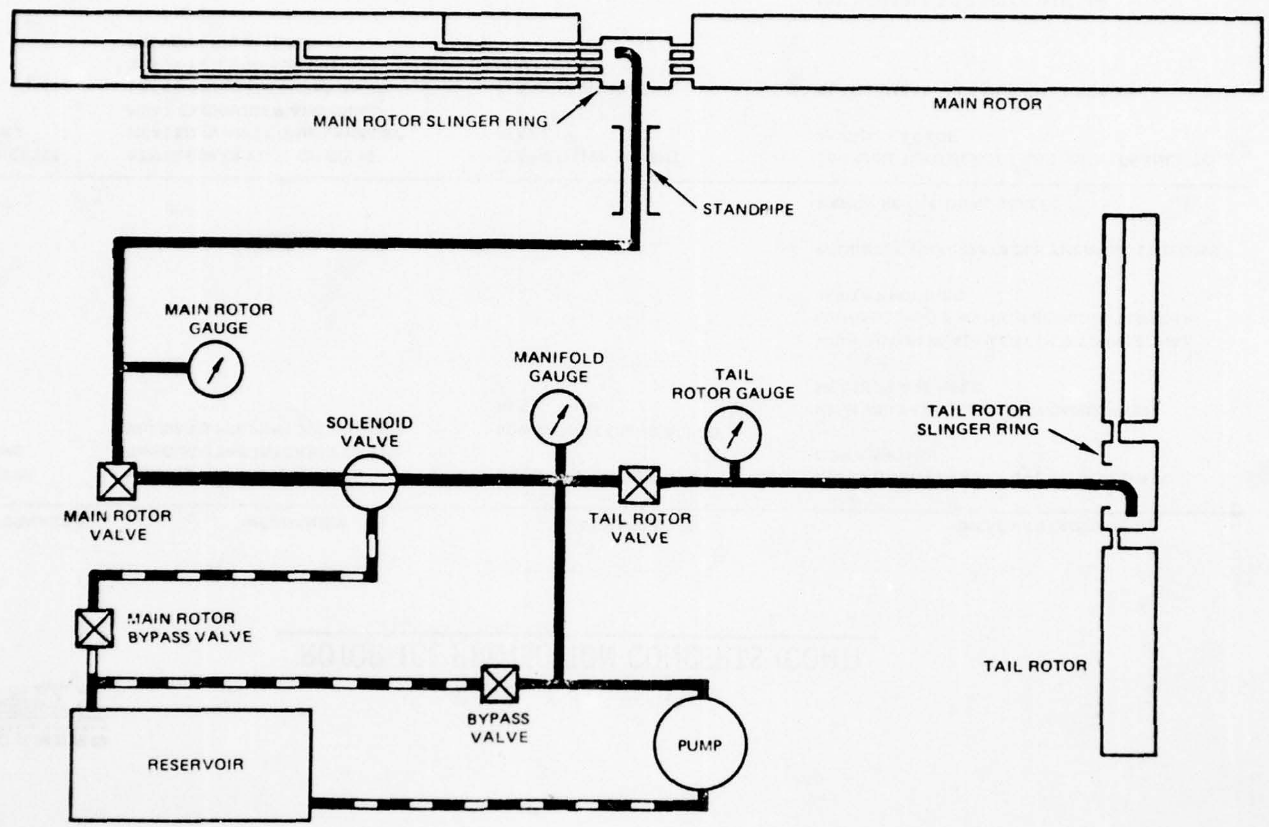
Figure 1

SCHEMATIC OF FLUID ICE-PROTECTION SYSTEM



The principle of operation is that the fluid mixes with water and the resultant mixture is eutectic. The weight flow requirements for alcohol are almost half that of glycol, but glycol is less volatile and more compatible with plastic hoses and paints. In general, the chemical system is simple and offers lowest production cost and is of a moderate to low weight. Its disadvantages include questionable protection performance due to poor deicing capability, unstable angle of attack effects (as the angle of attack increases on the blade, the ice catch tends to concentrate on the lower surface, but the fluid flows to the upper surface), and the need to have a fully wetted surface to insure good ice removal. Other disadvantages of the chemicals system are high vulnerability to battle damage due to relatively extended exposed area (reservoir and supply lines), need for field logistics, and the limitation on protection time due to reservoir size.

The second type of system seriously considered is shown schematically in Figure 2. This system fundamentally uses a liquid transport loop within the rotor blade that would interface with the engine exhaust gases as the primary heat source. Several methods were considered to accomplish the heat transfer. One would be as shown in the figure: a heat exchanger in the engine exhaust system and then pumping the liquid between the rotor system and the heat exchanger through a seal in the rotor hub. The second variation studied considered routing of the required quantity of the engine exhaust gas up to the hub and having a rotary type heat exchanger mounted on the hub. The liquid would be confined to the rotor system, and there would not be a requirement to transfer liquid across the seal. The third variation involved integrating the fluid heat transfer system with the engine transmission lubrication and heat rejection system. The liquid transport system with its efficient heat transfer properties, is the only one which offers the possibility of achieving a thermal fully evaporative anti-icing system on the rotor blades, because of the high heat transfer requirements for auto-icing (related to deicing). It does, however, have some areas of doubtful reliability and would require extensive development. It is also extremely susceptible to battle damage and may cause serious rotor blade vibration problems in the event of a system leak. Consequently, while this system has the potential of being the minimum performance penalty type of ice protection system because the heat that is used for the system is available virtually free, the probability for achieving a system of satisfactory service characteristics and reliability is not very high.

The third type of system considered is the electrothermal cyclic

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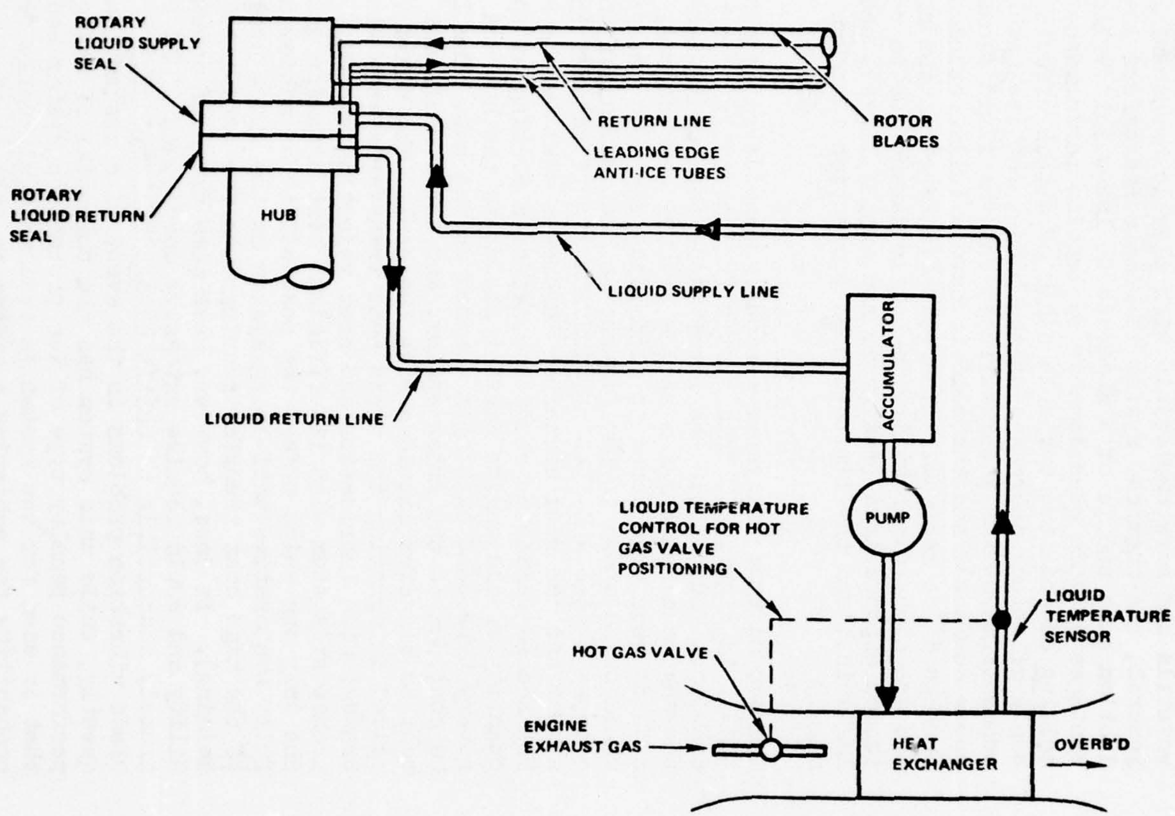
Figure 2

LIQUID ANTI-ICED BLADES UTILIZING ENGINE EXHAUST HEAT



deicing system, wherein the rotor blade leading edge area would be electrically heated on a transient basis and the heated area would be divided into a number of sequentially heated sections so as to minimize the electrical power consumption.

There are two possibilities relating to the configuration of the cyclic zones: one would be to divide the blade spanwise into a number of sequential sections starting with deicing at the tip and proceeding inboard to the root; the second possibility is to divide the blade chordwise, starting the deicing on the upper surface and proceeding forward to the leading edge and thence around several zones on the lower surface. Both of these approaches are illustrated in Figure 3. The spanwise elements with the chordwise shedding (the latter method) offers the simpler wiring system since all of the power leads can be concentrated at the root end and grouped together for collection. The configuration of chordwise elements in the spanwise shedding, on the other hand, requires the running of the various power wires along the blade as far as the tip. This approach, however, offers the possibility of providing a spanwise variation in power intensity to compensate for the variation in heat transfer properties due to aerodynamic heating along the blade. Further, the direction of ice removal is spanwise due to the centrifugal force effects, so that this is a more natural method of deicing than the chordwise shedding. Past electrothermal deicing attempts have mainly utilized the chordwise shedding system on the basis that the simplicity in the wiring outweighed the possible advantages of reduced power requirements. It is now believed, as a result of the current availability of flat wire and braiding, that the power leads can be incorporated conveniently into the rear of the deicer boot on the upper surface.

The three foregoing systems were compared during the study. One of the comparison parameters is the gross takeoff weight penalty for each of the systems (determined for a variety of vehicle sizes) accounting for the fixed weight of the system and the performance penalties associated with the system (the electrical power requirements and drag increase between shedding cycles in the case of the deicing system, the effect of heat exchanger on the engine back pressure and performance in the liquid-heated system, and the drag increase due to incomplete ice removal of the freezing point depressant system). This comparison is shown in Figure 4. The conclusion of this weight comparison is that the liquid-heated system, except for the smallest aircraft, has the highest takeoff weight penalty and that the penalties of the chemical system and electrothermal systems, depending upon vehicle size, are some-

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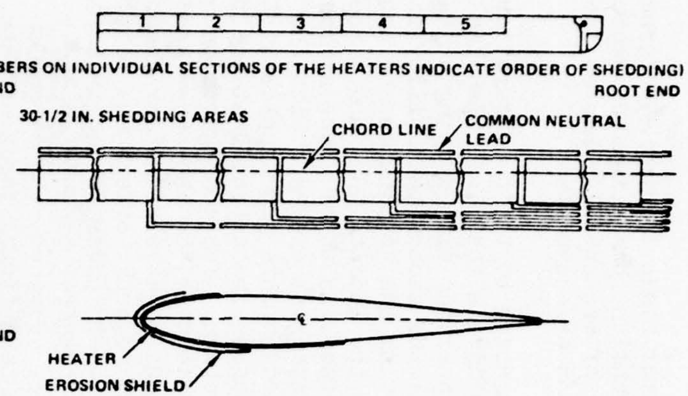
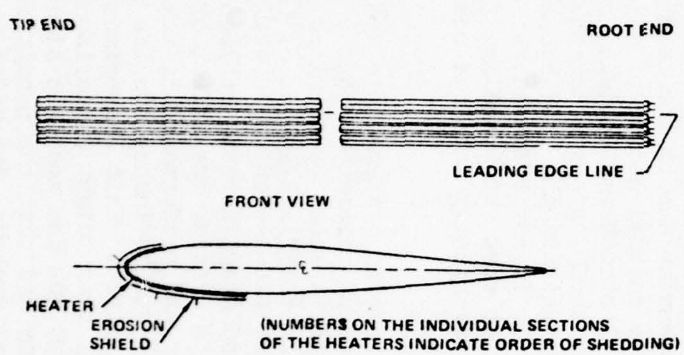
Figure 3

ELEMENTS FOR CHORDWISE/SPANWISE SHEDDING



CHORDWISE ELEMENTS FOR SPANWISE SHEDDING



SPANWISE ELEMENTS FOR CHORDWISE SHEDDING

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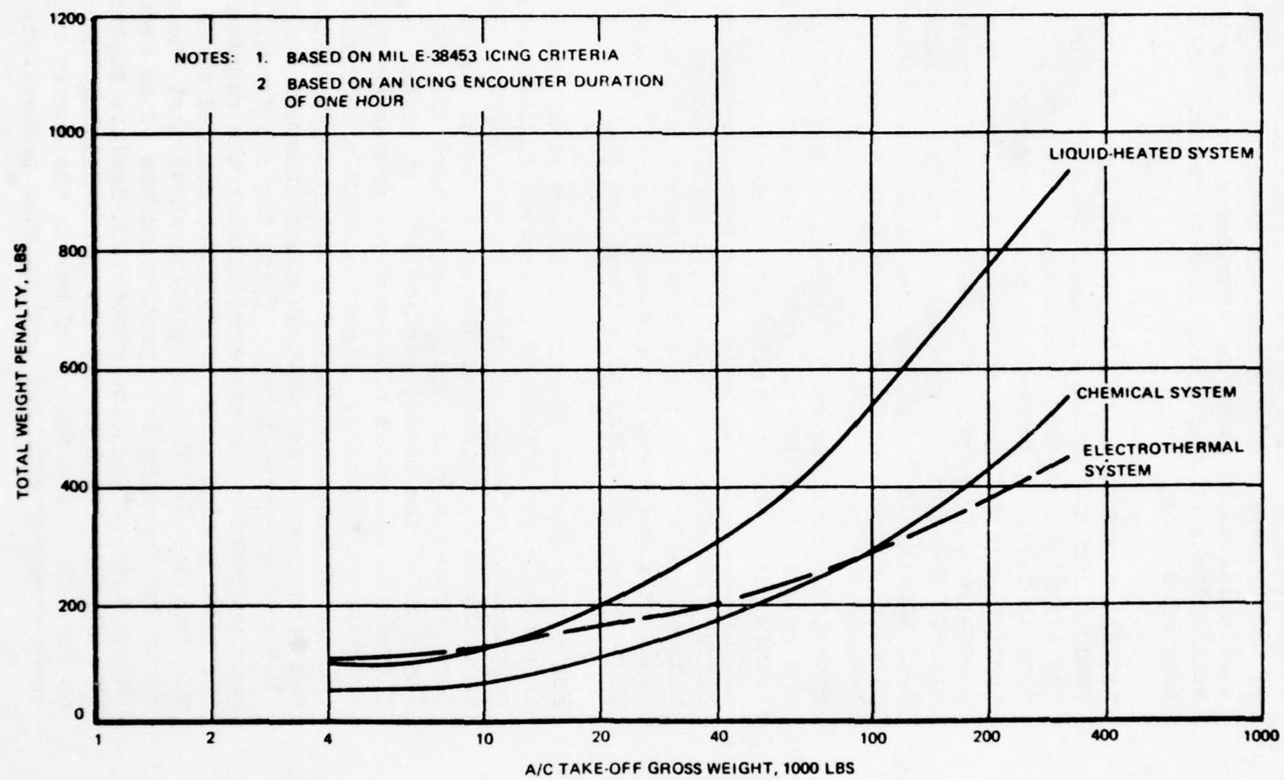
Figure 4

COMPARISON OF AVERAGE WEIGHTED PENALTIES FOR FLIGHT THROUGH ICING



what similar. On Figure 4 there is a major premise of a one-hour icing encounter duration. As indicated above, the chemical system requirements are a function of time spent in icing; thus, the weight penalty would be a significant linear function of time. If the criterion of the maximum aircraft endurance time for the particular missions being considered is used, the chemical system weight looks substantially less attractive relative to the electrothermal system.

Weight, however, is not the only figure of merit in comparing the various ice protection systems. Another figure of merit would be the effect on vehicle performance. Figure 5 shows such a comparison in terms of percent of cruise power required for ice protection. This figure shows that the liquid heated system has the lowest penalty because the back pressure effect is relatively minor. The chemical system, due to the allowance for a permanent drag penalty in icing, has the highest performance penalty, particularly for aircraft gross weights exceeding 10,000 pounds. An interesting factor which can be obtained from both Figures 4 and 5, especially from Figure 5, is that as the vehicle size decreases, the relative penalty for rotor blade protection rises. If these penalty data were replotted as a function of the percent of takeoff gross weight, it would show that the smaller the vehicle the higher the percent penalty for ice protection. The smaller vehicles tend to have the highest sensitivity to icing and the smallest power margins; consequently, being most in need of ice protection, are least able to afford it.

For electrothermal ice protection, a study was also performed to investigate the effect of protection against freezing rain as compared to supercooled water droplet clouds. When considering freezing rain, which has droplet sizes of 500 to 1000 microns, the large droplets will impinge or run back all the way to the trailing edge of the blade, particularly at higher angles of attack. Therefore, the whole blade must be protected to eliminate the freezing rain hazard. Figure 6 shows a comparison of the electrical energy requirements for a continuously heated anti-icing system (the upper line) with an electrothermal cyclic deicing system (the center dotted line). These two requirements for freezing rain protection are compared against the requirements for heating the leading edge area just for supercooled cloud conditions (the dot/dash line). This comparison shows that the anti-icing requirements for freezing rain protection are of an order of magnitude larger than for deicing protection against supercooled clouds. It also shows that even with cyclic deicing there is a difference on the order of a factor of two in the energy requirements between freezing rain and leading edge protection, particularly

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Figure 5

COMPARISON OF SHP REQUIREMENTS FOR FLIGHT THROUGH ICING

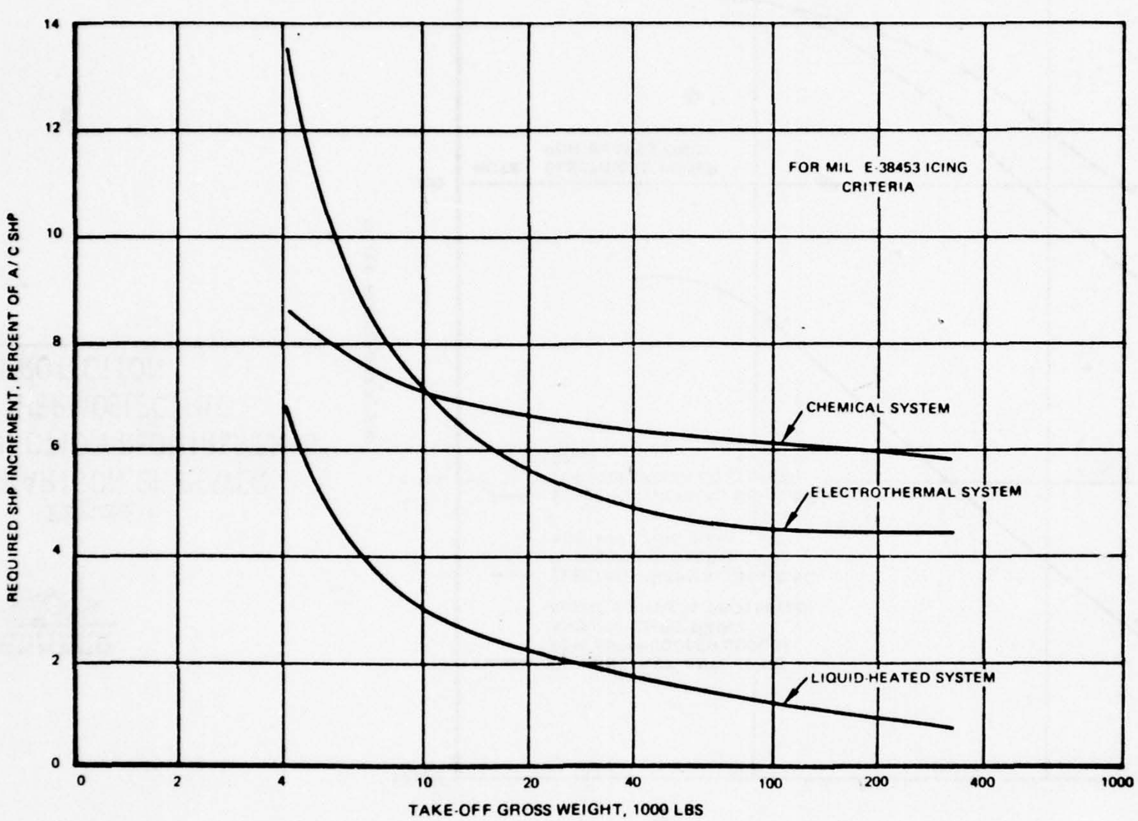

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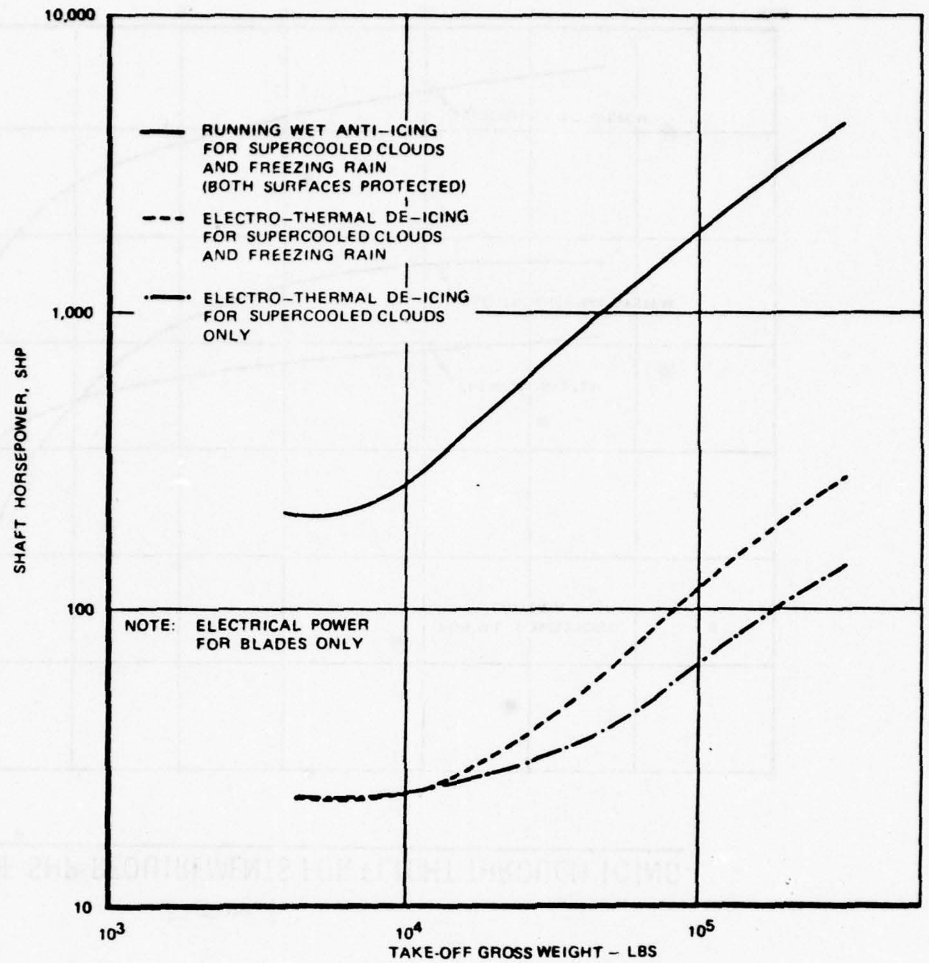
Figure 6
COMPARISON OF POWER EXTRACTION REQUIREMENTS FOR THERMOELECTRIC ICE PROTECTION



for the larger vehicle sizes. Considering the limited extent of freezing rain conditions, the relatively low probability compared to supercooled droplets, and the fact that evasive action can usually be taken, there is no justified rationale in designing an ice protection system that would fully cope with the freezing rain problem.

ELECTROTHERMAL DEICING TECHNOLOGY

As indicated above, the technology studies and tradeoffs resulted in a preference for an electrothermal cyclic deicing system for the leading edges of the rotor blades. Commensurate with supercooled cloud protection requirements, the coverage extends to approximately 25% chord on the lower surface of the blades, and to 10% on the upper surface of the blades.

There are two key components that are associated with the success of system design and operation. One of these involves development of a good control system for managing the power supply, distribution, and timing to the heating elements, and the second, of course, is the heating element itself. Figure 7 shows a simplified schematic of a possible control system that could be used for a blade deicing system. It is postulated that the basic electrical power system for rotor blades would be an ac system wherein all three phases are fed into each blade and that three-phase power is also fed into each heater section.

Over the years, experience in testing at the NRC spray rig has indicated that 1/4 inch of ice buildup at the leading edge of the blades is acceptable between deicing cycles. The actual buildup that would be experienced is a function of icing severity and, as shown in Figure 7, an ideal ice protection system would have a method of modulating system performance with icing severity (hence, the LWC severity meter input into the controller). With a fixed heat flux per heating element, that is a fixed watts per square inch value, the required "on" time would be a function of ambient temperature, and an ideal system would have its power "on" time modulated with OAT. There is a disadvantage in overheating a system; i.e., in allowing the temperature of the surface to rise in excess of 32 degrees before shutting off. This problem is associated with runback. It takes a finite amount of time for the system to cool down, and, with the interface temperature above freezing, the ice that is caught on the heated area proceeds to run back and refreeze in the unheated area aft of the deicer. Experience has shown this to be a particularly important factor in the successful operation of deicing systems for fixed wing aircraft. There is, however, limited information as to the effect of runback or even its occurrence on helicopters since it is more

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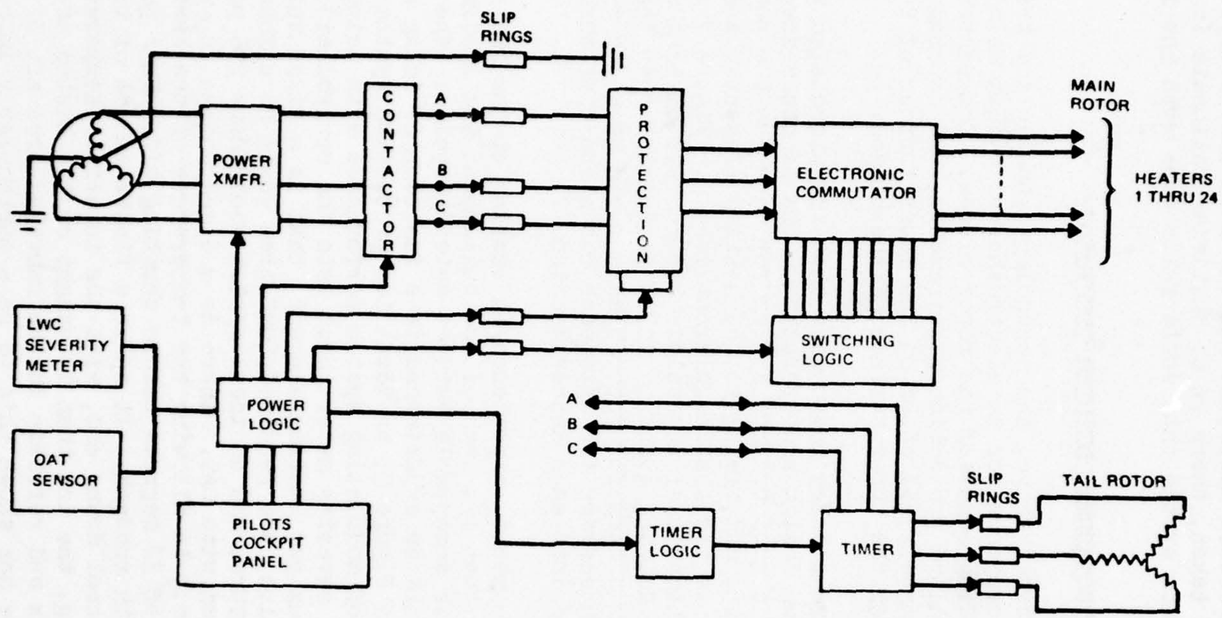
Figure 7

DEICING CONTROL: SIMPLIFIED BLOCK DIAGRAM



likely that the water will run spanwise along the heated area rather than chordwise. Thus, testing is really required to identify what happens.

There are a number of ways to transmit power from the ship's electrical system into the rotor blade. Studies in this program have shown that slip rings offer the minimum weight systems, and experience with propeller deicing systems indicate that these can be highly reliable. Since the rotor blade would be divided into a number of segments, a method of distributing and sequentially managing the power to each one of these sections must be considered. One way to do this would be to use a master commutator in the aircraft, with the number of slip rings on the rotor mast equal to the product of cyclic sections and the number of phases. For example, an 8-segment system would require 24 slip rings for the deicing system. The alternative to this would be to have a commutating system in the rotor head and simply deliver the three-phase power with three power slip rings supplemented by additional control slip rings to the commutator in the rotor.

The power management problem with the tail rotor is somewhat simplified compared to the main rotor. Usually, the protected area for the tail is much smaller than that for the main rotor, and the tail rotor can be deiced with one segment at less power than required for one of the segments of the main rotor system. Hence, Figure 7 shows only a single heater element system for the tail rotor.

Considering the large number of slip rings that permits doing all of the switching on the fuselage side vs. commutating in the rotor head, a tradeoff study suggests that it would be preferable to reduce the number of slip rings and do the switching in the rotor head. To accomplish this function, an all solid state system, an electromechanical system, or a combination (hybrid) of solid state and electromechanical commutating switches could be considered. In switching relatively large amounts of power (for example on an aircraft the size of the UH-1H this is on the order of 50 amps per phase), electromagnetic interference (EMI) has to be seriously considered in the design of the system. Electromechanical systems tend to be heavy, have significant EMI problems, and if great care is not exercised in their detail design, there is a limited contact life due to arcing. A solid state switch appears to have the possibility of overcoming these disadvantages; but a solid state system has other disadvantages. The amount of current going through the switch results in significant heat dissipation problem in the commutator assembly due to the relatively low efficiency of the components. Also, the space and volume required for the solid state switch is directly proportional to the

number of heating elements into which the blades are divided. Currently, solid state units are not available in production quantity with the required mil standard quality. In addition, detailed layouts of a solid state switch for a rotor sized for the UH-1H showed that the space would be limited to about 5 or 6 three phase elements per blade. The third alternative - the hybrid system - would utilize solid state components for controlling an electromechanical stepping switch and would also use a solid state master switch to eliminate the EMI problem. While the mechanical stepping switch in the hub is moving from one position to another, as shown in Figure 8, the solid state master switch in the helicopter would be closed and would close during the zero current phase of the ac wave (thus eliminating EMI).

One of the considerations in selecting a power distribution system is the number of sections into which the blade heater will be divided. The choice is not obvious. As the number of segments increases, the amount of wiring and system complexity increases. On the other hand, the electrical power requirements are reduced. Conversely, the fewer the number of segments the higher the generator weight and power. And, if as indicated, a solid state controller system is utilized, the weight of the switch gear is proportional to the number of sections. (The electromechanical stepping switch, however, tends to be more independent of the number of vehicle segments.) A study, therefore, was conducted to compare the overall aircraft system weight using both solid state and hybrid systems as a function of the number of deicing elements. The results of this study are shown in Figure 9. This shows that the system weight tends to become relatively flat (for a 10 to 15,000 pound vehicle) above about 5 segments and that the hybrid controller has a lower weight than the solid state system due to the heat sink dissipation requirements of the solid state system. Based on the foregoing considerations, it has been concluded that the hybrid type of deicing controller offers the best potential for the deicing system.

The most critical component, in terms of the current state-of-the-art of deicing system technology, is the actual heater element assembly applied to the blade surface. The unsatisfactory development status of this component is what has led to the current deficiency in rotor blade deicer technology. Past experience with both fixed wing aircraft and rotary wing aircraft has indicated a number of problems which are summarized on Table II. The deicer boot consists of (1) an erosion shield, (2) a forward dielectric, (3) a heating element, and (4) a rear dielectric between the deicer and the blade. The requirements for a successful deicer assembly includes providing uniform heating over the whole surface area, with no spots either too hot or too cold. A

Figure 8

HYBRID SEQUENTIAL POWER SWITCH

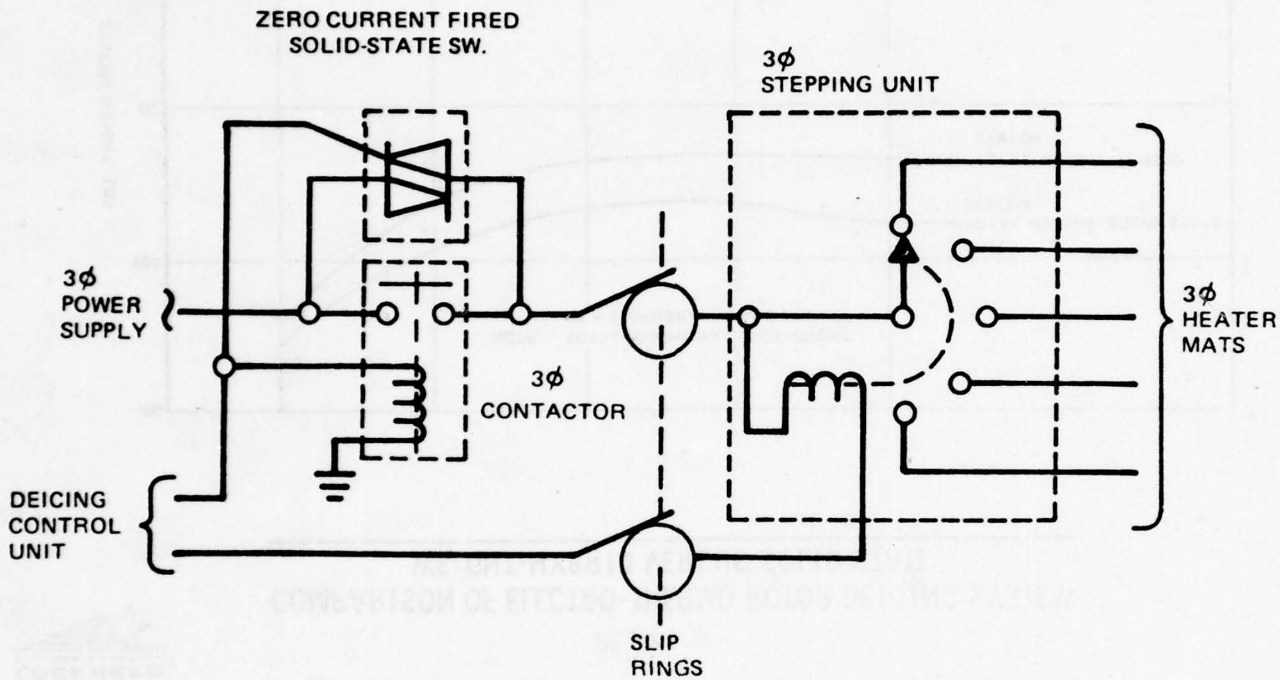


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Figure 9

COMPARISON OF ELECTRO-THERMO ROTOR DEICING SYSTEM WEIGHT-HYBRID VERSUS SOLID STATE

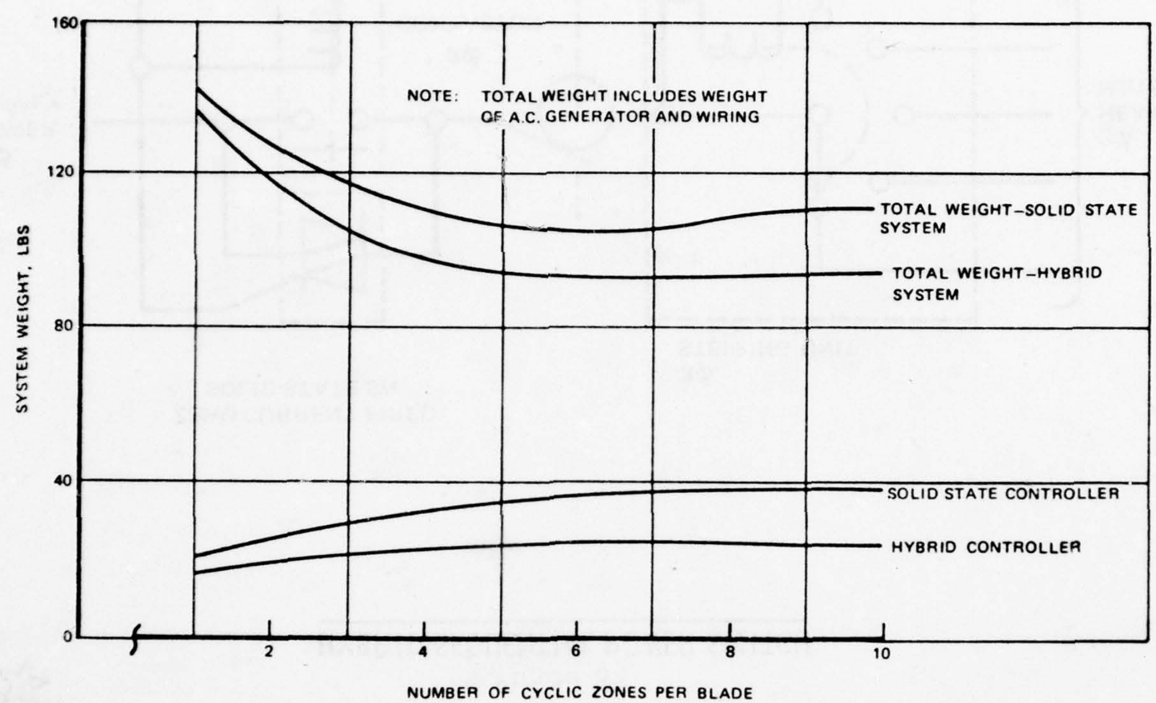

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Table II

PROBLEM SUMMARY ELECTRICAL DEICER SYSTEMS

PERFORMANCE REQUIREMENT	POTENTIAL SERVICE PROBLEMS	PROBABLE CAUSES
HEAT UNIFORMITY	HOT SPOTS, COLD SPOTS	VARIABILITY IN HEATING ELEMENT RESISTANCE. VOIDS IN ADHESIVE BONDLINES AND INSULATION LAMINATES. MAY OCCUR IN FABRICATION OR BY DELAMINATION IN SERVICE. VARIATION IN THICKNESS OF ADHESIVE BOND LINES AND INSULATION LAMINATES.
PREVENT MALFUNCTION OF HEATERS	BROKEN ELEMENT	THERMAL STRESS FATIGUE, APPLIED STRESS FATIGUE, LOW STRAIN CAPABILITY, AND DUCTILITY OF ELEMENT.
	SHORT CIRCUIT IN ELEMENT	STRESS FATIGUE OF BONDLINES AND INSULATION CAUSING VOIDS AND DELAMINATION IN CONJUNCTION WITH MOISTURE ENTRY AND DISPLACEMENT OF ELEMENTS UNDER DYNAMIC LOADING.
	SHORT CIRCUIT-ELEMENT TO EROSION SHIELD OR STRUCTURE	UNDETECTED FABRICATION FLAWS OF MARGINAL BOND STRENGTH IN LAMINATES MAY BE CONTRIBUTING FACTORS.
DAMAGE TOLERANCE AND EASE OF REPAIR	DAMAGE ELEMENT OR INSULATION	PHYSICAL DAMAGE OR ELECTRICAL SUPPLY MALFUNCTION.

second requirement is that there shall be no delaminations or bond failures. Additionally, the deicer assembly must be free from short circuits or open circuits; and, in the event of damage, it shall be conveniently reparable.

Such a system has existed on the fixed wing P-3 ASW Navy Patrol Aircraft for the past 15 years. One deicing system is used for the empennage, and a second deicing system is used on the propellers and spinners. These designs have been used as the point of departure for the system being developed as an advanced helicopter deicing system. A development program related to modifying it for helicopter application was accomplished during the past fall and spring. This entails the substitution of stainless steel or nickel for the aluminum erosion shield, on the P-3, improvement of dielectric material that was originally developed 15 years ago, and accounting for the unique stress requirements on the erosion shield due to the rotor loads. An ideal heating element would be a thin continuous metal sheet that would offer an instantaneous temperature rise to the required level. The departure from this ideal results in an increase in heating element temperature over a discrete time period. Also, there will be, depending upon the materials of the system, a non-uniform surface temperature, with the hottest temperature directly over an element and coldest midway between elements. In addition, as the amount of dielectric material is increased between the heating element and the surface clading, the temperature differential between the two must of necessity increase. These factors result in a slower heating time and more heat storage in the system, thus increasing the extent of runback. Any heat storage increase also represents wasted energy. Voids in the bond between the dielectric material and the cladding, or between the heating element and the dielectric are absolutely unacceptable. These create temperatures sufficiently high to cause dielectric or heater element failure and also cold spots on the surface. A quality control program must be implemented during and at the completion of the heater assembly so that the uniform heating can be assured and demonstrated by test. The design could have either a nickel or a stainless steel erosion shield; for production purposes a nickel erosion shield with a maximum thickness of 0.030 at the leading edge tapered to 0.010 at the aft end of the erosion shield would be appropriate for both main and tail rotors. However, in this prototype test program, there was a very large cost and schedule penalty for using nickel (particularly on the main rotors), and a decision was then made to use stainless steel erosion shields for the main rotor blades and nickel for the tail rotor. Nickel, while more costly, offers the best erosion resistance and is expected to be 3 times more cost effective in production. The dielectric materials selected represent a combination of pre-preg epoxy with glass cloth mixture with

nitrile phenolic adhesives, and the heating element is an etched foil type of design very similar to that which has been successfully applied on the P-3. The etched foil design, which is only .005 thick, offers a very good approach to getting the ideal planar surface and gives excellent control over the power intensity. By properly zigzagging the pattern, the design can accommodate normal thermal expansion. Since the individual heating elements are about 0.2 inches wide, it is possible to effect repairs. Field repairs on small damage areas can be made, and larger repairs can be and have been accomplished at overhaul depots. Repairs are made by cutting into the cladding and sectioning in a new piece of heater element.

It is possible to use dielectric thicknesses of as little as .005 inches between the heating element and the cladding. A thickness approximately double that is being used to provide a more rugged design. Based upon these layer thicknesses, an estimate has been made of the power intensity required as a function of ambient temperature for two stations along the rotor blade, and this is shown in Figure 10. As indicated earlier, the power requirements are a function of distance along the span as a result of the aerodynamic heating. For example, the temperature rise in dry air at the blade tip is approximately 48 degrees F. This rise in the aerodynamic heating along the span toward the tip alleviates the heating requirements in this direction. Consequently, the time required to reach the critical 32 degree interface temperature with a given power intensity and at a given ambient temperature is reduced towards the tip. This is clearly illustrated in Figure 10 which compares the time required to reach 32 degrees F at the 52 percent span station and at the 82 percent station.

The probability studies relating to the likelihood of occurrence of icing established that the minimum design ambient temperature for system design is minus 4 degrees F (which coincides with the AV-E-8593 Army spec for engine ice protection). Therefore, this ambient temperature was used in plotting the power density required as a function of the distance along the span (Figure 11). Each flat portion indicated is associated with one of the cyclic zones.

UH-1H MODIFICATIONS AND PLANS FOR FLIGHT TESTS

Once it has been concluded that the cyclic thermal electric deicing system presents the most promising concept of rotor blade ice protection, the practical aspects of incorporating such a system into the UH-1H helicopter have been considered. The basic UH-1H helicopter does not have any rotor blade deicing sys-

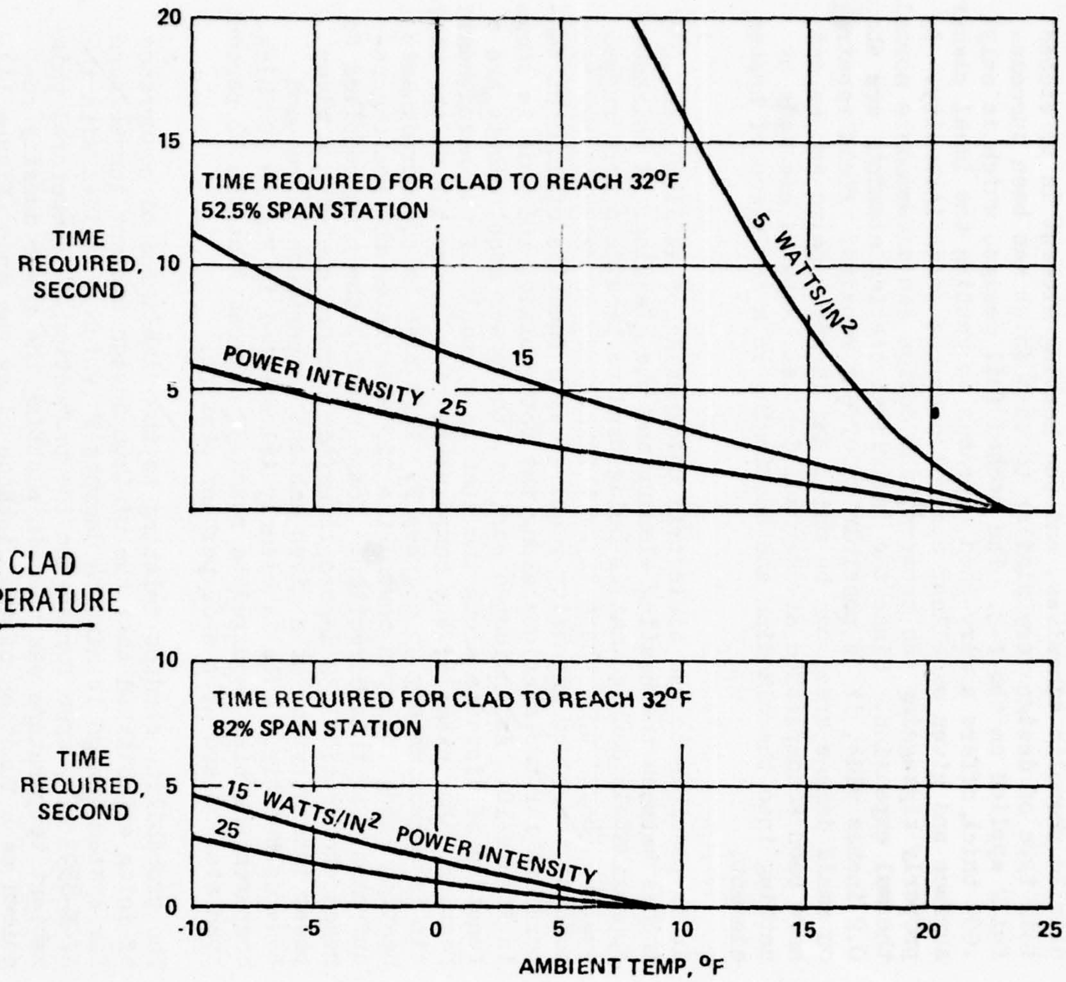

Figure 10
TIME REQUIRED FOR CLAD TO REACH 32°F TEMPERATURE

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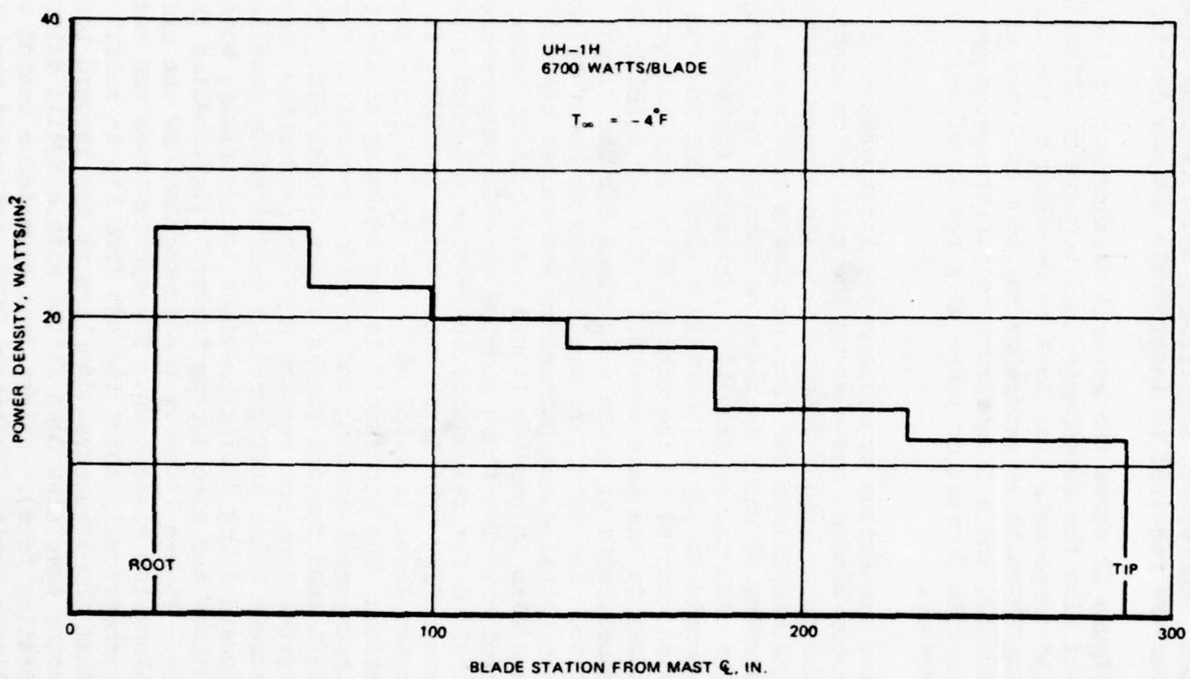
Figure 11

MAIN ROTOR BLADE SPANWISE POWER DENSITY DISTRIBUTION



tem, and, also its electrical generating capacity is not adequate to accommodate the requirements of the cyclic thermal electric system. Therefore, significant modifications of the electrical system are required to incorporate the new deicing system.

Figure 12 shows the general arrangement of the UH-1H helicopter and also the components that require modifications, as well as the components that have to be added to the aircraft. The existing 28 volts dc generator has been replaced with a 20 kva ac generator, and a transformer rectifier system has been installed to provide 28 volt dc power as a result of deleting the 28 VDC generator.

Ice protection provisions are incorporated on the main and tail rotor blades, and also on the main rotor stabilizer bar. While the rotor blades feature a cyclic electrothermal deicing system, the stabilizer bar incorporates a continuous heater, the power intensity of which is based on running wet surface requirements. The distributor assembly, including the power control system, is mounted on top of the main rotor. The instrumentation package is also mounted on the rotor. Slip rings are incorporated to deliver power to the main and tail rotor for deicing the leading edge surfaces, with separate slip rings for the flight test instrumentation. Ice detectors are mounted on the fuselage to insure that the actual cloud parameters are compatible with predicted values and also to provide liquid water content value input into the controller. Two flush mounted outside temperature probes are installed for redundancy to provide a second input into the cyclic controller (the first input being the LWC), so that the proper power level is supplied to the cyclic zones for the required duration. The windshield is heated using an electrically conductive transparent coating and provides protection for both the right and left hand forward panels of the windshield. The engine induction system does not require any ice protection, because the pleated-screen inlet configuration exhibited an excellent tolerance toward icing during previous icing trials, both during hover at Ottawa and also during forward flight behind the CH-47 Army Tanker. Figure 13 shows the arrangement of the cyclic heater zones along the blade span. The cyclic zones are energized in sequence symmetrically about the hub from tip to root. Figure 13 shows that the structural doublers at the inboard portion of the main rotor have also been fitted with a cyclic deicer (requiring one heating zone). Flight Test experience behind the CH-47 Tanker in Alaska during September, 1973, indicated that ice on the doublers affects the auto-rotative capability of the helicopter and, therefore, heater elements on this portion of the span are necessary.

The cockpit control panel for the experimental UH-1H helicopter

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Figure 12

GENERAL ARRANGEMENT - UH-IH DEICING SYSTEM

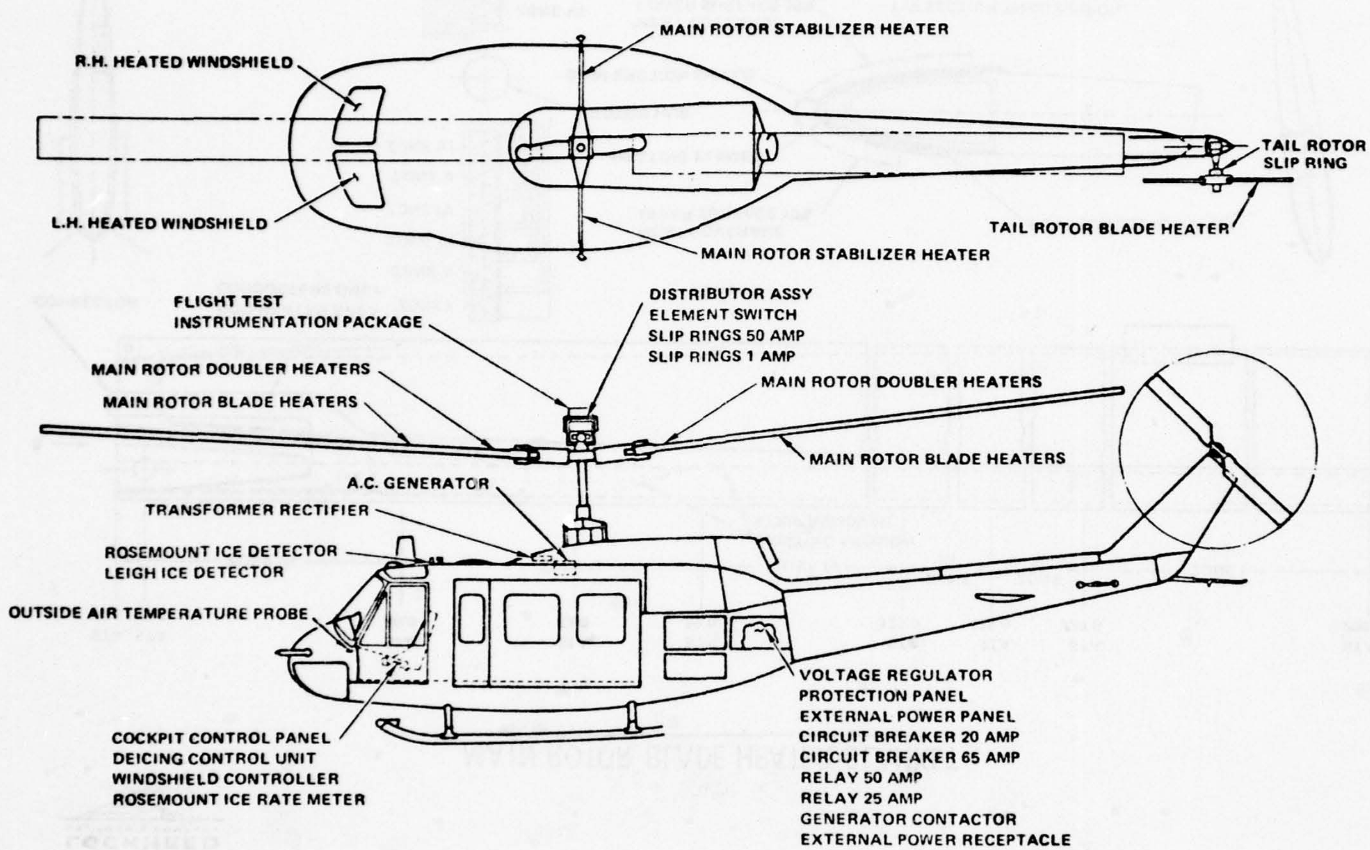

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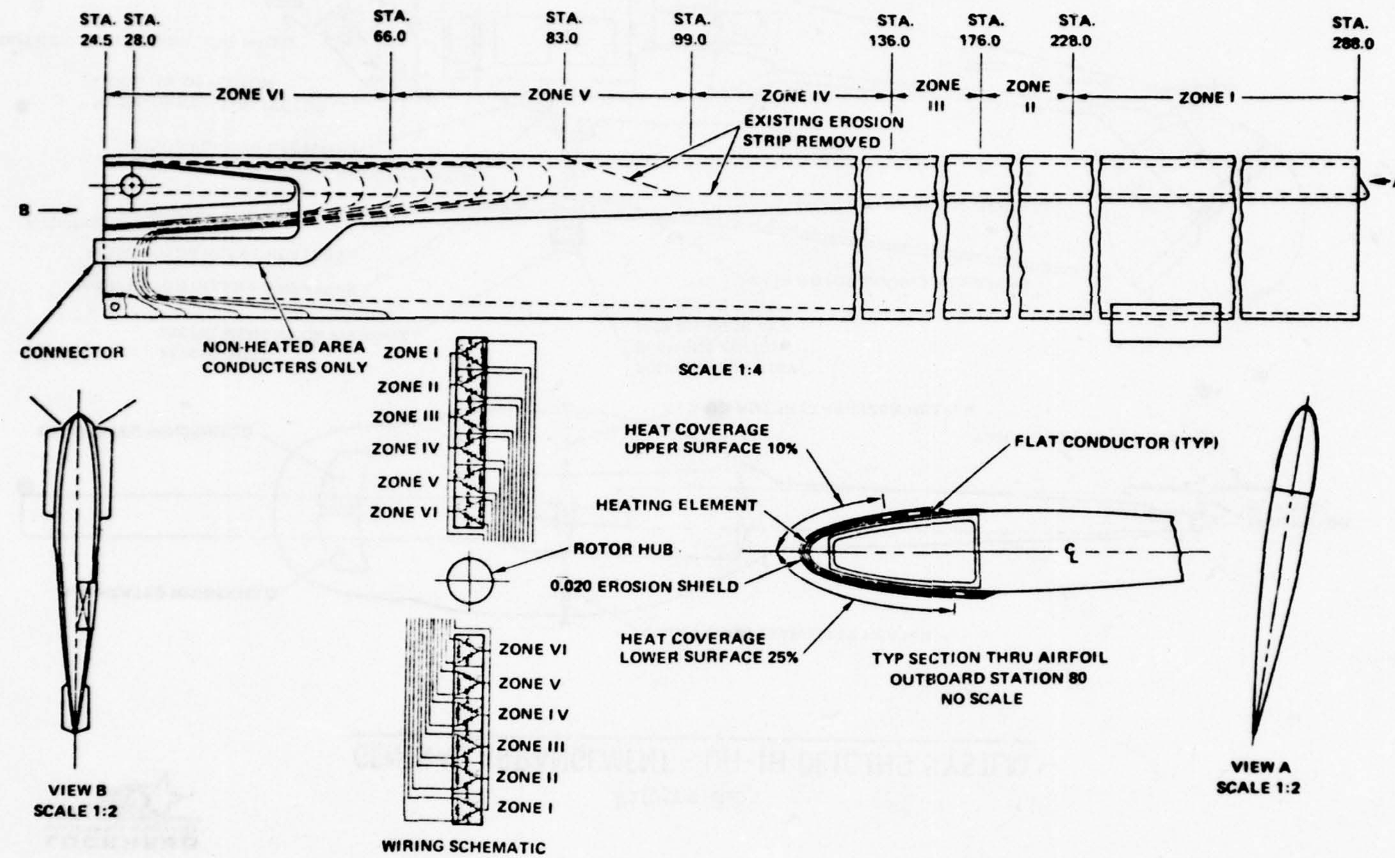
Figure 13

MAIN ROTOR BLADE HEATER BLANKET



(Figure 14) features the ambient temperature indicator and also an LWC indicator. The rotor blade control system which has been configured for the test aircraft provides a very complete degree of operational flexibility - more complete than would be incorporated into a production design - in order to evaluate during flight test the best method to be utilized for production. Two modes of operation are provided. There is a completely automated mode wherein both the ambient temperatures and the liquid water content values are automatically fed into the control system to control the energy level and cycle time. These two parameters, the liquid water content and the ambient temperature, serve as inputs to vary: (1) the heating "on" time, (2) the cycle dwell ("off") time, and (3) the heater energy levels. In addition, there is a semi-automatic mode wherein the pilot can input the icing severity (i.e., light, moderate and heavy) and the ambient temperature and regulate the power and on-time for the heater elements.

To modify the heater power density (watts/sq. inch), there are three generator voltage levels (160, 200 and 230 volts) which are automatically adjusted in accordance with the three discrete LWC severities. The OAT control operates such that the lower the OAT, the longer the energy "on" time. The cycle dwell time is reduced as the icing condition becomes more severe.

The pilot-operated, simpler semi-automatic mode will permit a manual adjustment during icing flight test trials to determine optimum deicing schedules for the automatic controls. To this extent, it will serve to resolve any discrepancies in the control requirements between the design and actual icing flight test conditions. Once these differences have been identified, the appropriate adjustments can be made to the controls, using the built-in manual adjustment provisions. By means of these adjustments, the energy quantums applied to individual zones can be changed ± 50 percent by varying the "on" time for the individual zones. Also, there are provisions for adjusting the cycle dwell (off) period.

The aircraft's three-phase generator is being run in an ungrounded neutral mode. By doing this, the prospects of high line-to-ground fault currents are eliminated and, therefore, the possibilities of structural damage to the rotor blades by current arcing. While the new deicing system is essentially free of the problems of line-to-ground faults; if such a fault were left undetected, then a second fault on another line (phase) would be tantamount to a line-to-line short. To detect this hazard, a ground-leakage warning is provided and this results in the lighting of an indicated light on the pilot's panel if a single line-to-ground fault occurs.

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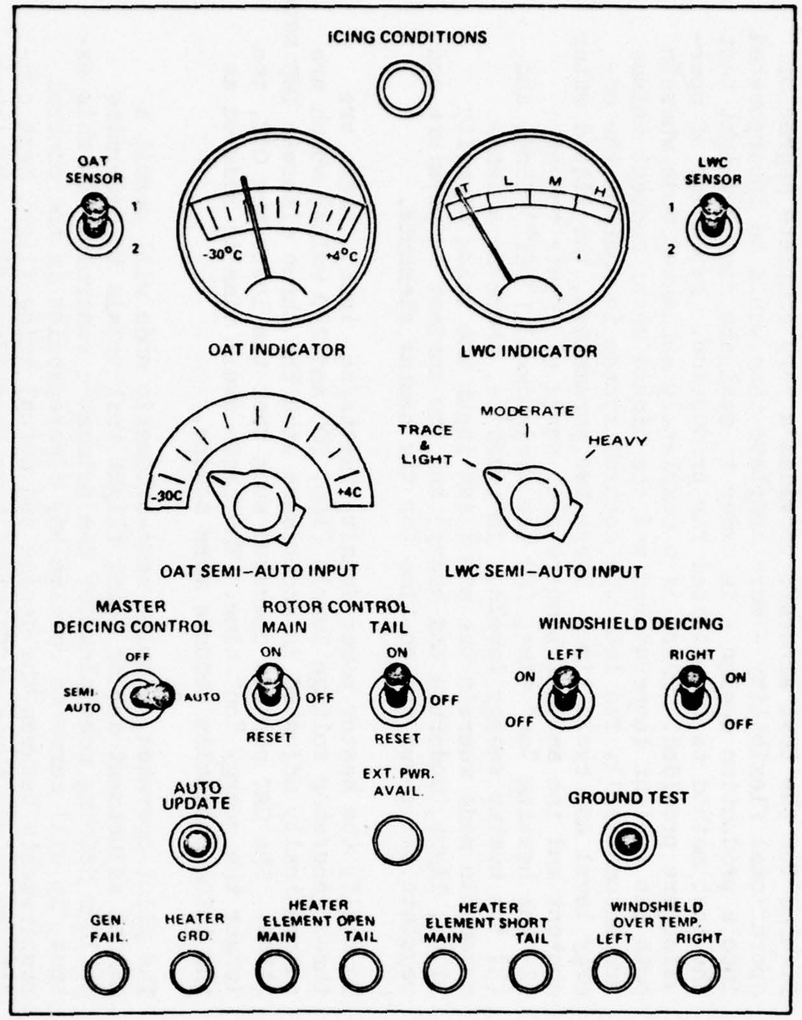
Figure 14
COCKPIT CONTROL PANEL



In addition, it is an objective of the system to provide separate protection to the main rotor, the tail rotor and the stabilizer bar heating systems. Each of these sub-systems, therefore, have individual relay controls and protection logic capable of detecting fault problems in these individual systems. Current transformers, therefore, monitor the input power to each sub-system and discriminate any condition of overcurrent or current unbalance. If line-to-line shorts occur, this would result in an overcurrent and a current unbalance. The logic responding to this acts to latch it out in such a way as to open the respective line relay.

In the case of an open phase or a break in one of the three-phase heater elements in any main rotor blade heater section, the logic performs differently and commands the stepper switch to pass quickly through the position (supplying the faulty element) and then continue the cycle in the normal sequence. This means that deicing capability is lost not only on the heater element with the fault, but also its opposite symmetrical heater elements. This control technology allows reduced deicing capability to be maintained on the main rotor and for flight to be continued, with a minimum impact on any unbalanced moments of the rotor system.

Overcurrents represent a more significant hazard than open circuits, and, for this reason, the appropriate sub-systems are isolated if such overcurrent faults are detected. If the overcurrent/unbalance fault occurs in the supply lines to the deicing system, the master relay is opened to isolate the whole deicing system, with the exception of windshield deicing.

The pilot is also able to monitor visually the status of the deicing system by means of magnetic indicators which are also provided on the deicing controller located in the aft end of the central console in the UH-1H. Ground testing is accomplished by means of a press-button on the pilot's control panel and this initiates a fast 2 second scan of all heater blade heater elements. If any fault is detected during this fast scan, it will be flagged on the deicing controller.

The flight test program which will be carried out starting next winter will have as its primary objective the demonstration of the foregoing deicing systems, with principal emphasis on the rotor system. Secondary objectives will be to determine the best type of system control (automatic, semiautomatic, or even a fixed cycle independent of icing severity), the need for an ice detector or liquid water content meter, and the power requirements for wincshield anti-icing. In addition, the airplane will undergo tests to establish its airworthiness and to verify functional

system operation. These tests would involve flight tests of the vehicle in dry air and in icing.

The flight test program will consist of four phases (Table III). The first phase will include exploration of the performance envelope of the helicopter to insure that the modifications that have been incorporated do not introduce any detrimental effects. These tests will be conducted in dry air and will also include a functional checkout of all the deicing equipment and data systems. After completion of the first phase of the flight tests in dry air, the aircraft will be ferried to Ottawa where it will first undergo icing flight tests in hover conditions. Upon completion of the icing tests in hover, it is planned that the aircraft will fly behind the CH-47 icing tanker for the evaluation of the ice performance of the ice protection systems during forward flight. If possible, further tests will be conducted during natural icing tests. It can be seen from Table IV that there is sufficient instrumentation planned for the UH-1H aircraft to monitor the structural and vibration characteristics of the aircraft and to monitor the power changes or changes in the flying qualities, if such are encountered. Furthermore, there is sufficient ice protection system measurements to enable verification of the effectiveness of this system which includes both blade and windshield ice protection. Of cource, all flight conditions including speed, altitude and engine torque, rpm, etc., will be monitored, as well as the major parameters of the newly installed generator system. A motion picture camera mounted on the top of the hub will enable to evaluate, at least qualitatively, the performance of the icing system of the main rotor blades.

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Table III

FOUR FLIGHT TEST PHASES

1. AIRWORTHINESS AND FUNCTIONAL CHECKOUT TESTS.
2. HOVER ICING SIMULATION TESTS.
3. AIRBORNE ICING SIMULATION TESTS.
4. NATURAL ICING TESTS

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Table IV

INSTRUMENTATION

- 19 STRUCTURAL LOADS/VIBRATION MEASUREMENTS

  4 MAIN ROTOR
  3 TAIL ROTOR
  12 VIBRATION

- 28 ICE PROTECTION SYSTEM MEASUREMENTS

  4 ICING CONDITION PARAMETERS
  10 ROTOR BLADE AND WINDSHIELD TEMPERATURES
  14 A.C. GENERATOR SYSTEM PERFORMANCE AND OPERATION

- 10 GENERAL TEST CONDITION MEASUREMENTS

  IAS, ALTITUDE, ENGINE TORQUE, RPM, ETC.

- 1 ROTOR HUB MOUNTED MOTION PICTURE CAMERA

ANTI-ICING, DE-ICING, DE-FOGGING CONSIDERATIONS FOR HELICOPTER TRANSPARENT ENCLOSURES

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INTRODUCTION

The ever increasing market for helicopters and the emphasis placed upon the need for newer and more modern aircraft such as the UTTAS has dictated that each component of the helicopter be more sophisticated and better than anything previously built by the industry. Developments in this direction are sure to continue with additional technology needed to meet the required service life of each component. The transparent enclosures of helicopters are becoming, along with the total helicopter, more sophistocated and thus there are more and more items which must be considered during the design phase of the windshields.

Of primary importance to the helicopter manufacturer is the ability of the primary windshields to de-ice and de-fog so that the helicopter retains an acceptable level of performance in severe weather conditions. In order to accomplish this mission, there are many items to be considered.

The first step in the development of a transparency is to determine the heat requirement for the atmospheric conditions that the helicopter will encounter. PPG has recently been investigating the heat requirement calculations using a computer program. A review of our findings to date and an analysis of this program is presented below.

HEAT REQUIREMENTS

As stated in the helicopter section of Federal Aviation Agency, Technical Report, ADS-4 (Reference 1), "the wide variety of flight modes and variations in windshield sizes, shapes and installations make generalization of windshield ice protection requirements virtually impossible." Thus, the amount of heat required to prevent icing is not only dependent on the aircraft speed, as indicated by MIL-T-5842A (Figure 1) but is dependent on several other important parameters. Other major parameters which can effect the amount of heat required are windshield height, the angle of installation and the distance from the

stagnation point to the windshield. After a brief description of the meteorological conditions which cause icing problems we will show the significance of these parameters as predicted by a computer program which is now available to the aircraft industry. The theory of this program is also presented for evaluation.

Meteorological Icing Conditions

Before we can efficiently determine the heat requirements for any aircraft we must first compare the aircraft operational envelope with the known meteorological conditions at which icing can occur.

Icing clouds generally fall into two catagories: stratus and cumulus. Stratus clouds generally exist at altitudes from sea level to 22,000 ft. and have a horizontal extent of 20 to 200 miles. Cumulus clouds exist at altitudes from 4000 ft. to 22,000 ft. and have a horizontal extent of 2 to 6 miles. Icing conditions which exist in stratus clouds are known as "continuous" while icing conditions which exist in cumulus clouds are known as "intermittent".

Envelopes for continuous and intermittent icing conditions that were gathered by NASA from natural icing conditions are presented in Figure 2 and Figure 3, respectively.

Atmospheric conditions which can effect the amount of heat required for a windshield include the following:

- liquid water content (LWC)
- effective droplet diameter
- temperature
- altitude

These icing parameters are inter-related as can be seen in Figure 4 for the continuous maximum icing condition and Figure 5 for the intermittent maximum icing condition.

These atmospheric conditions along with the physical characteristics of the aircraft, such as:

- speed
- windshield angle of installation
- windshield height
- distance from aircraft stagnation point to windshield

are considered in a computer program (Reference 2) which was written by George Letton as a graduate student at Ohio State University in 1972.

Computer Program for Calculating Heat Requirements

Mr. Letton approached the problem of windshield heat requirement analysis similar to methods used in the past, however by inputting a minimum number of aircraft characteristics we can quickly obtain a complete performance map of all possible heat requirements from his computer program. Those of us who have gone through heat requirement calculations by hand can fully appreciate the usefulness of this or a similar computer program.

Listed below is a summary of the equations and assumptions used in Mr. Letton's analysis.

The analysis is based on maintaining the outer surface at 35°F in a "running wet" condition.

The total heat flow, q_s, at the outer surface of an aircraft windshield can be expressed as the sum of four individual heat losses as follows:

$$q_s = q_1 + q_2 + q_3 + q_4$$

where

q_1 is the heat loss due to forced convection

q_2 is the heat loss due to the heating of the impinging water droplets (including consideration of kinetic energy available from drops)

q_3 is the heat loss due to evaporation of some or all of the impinging water droplets

q_4 is the heat loss due to radiation

Each of these four heat losses are analyzed as follows.

Convective Heat Loss (q_1)

The basic equation for the unit convective heat loss is expressed by:

$$q_1 = h_{avg} \; (t_s - t_{aw})$$

where

t_s is the outside windshield surface temperature, which is 35°F for this analysis

t_{aw} is the adiabatic wall temperature in °F and is expressed by:

$$t_{aw} = t_\infty + \frac{rU_L^2}{2g\, c_p\, J}$$

h_{avg} is the average convective heat transfer coefficient over the surface of the windshield in Btu/hr.-ft.^{2}-°F

r in the equation for t_{aw} above is recovery factor and can be expressed in terms of Prandtl number, Pr as follows:

$$r = (Pr)^{1/3}$$

Assuming that turbulent flow exists over the windshield, Mr. Letton used this equation for the convective heat transfer coefficient (turbulent flow over a flat plate)

$$h_{avg} = 0.0296 \frac{k}{X} Re_x^{0.8}\, Pr^{0.333}$$

where

X is the distance from the stagnation point to the center of the heated windshield area in feet

Re_x is the dimensionless Reynolds number at the center of the heated windshield area

Since the analysis is valid for turbulent flow only, the value of Re_x should be greater than $5x10^5$. The equation for the dimensionless Re_x is expressed as

$$Re_x = \frac{\rho U_L\, x}{\mu}$$

Mr. Letton assumed that the local velocity over the windshield, U_L, is equal to the freestream or aircraft velocity, U_∞.

Heating of Impinging Water Droplets (q_2)

Mr. Letton used the following equation for the unit heat loss due to the heating of water droplets impinging on the windshield heated area, which takes into consideration the kinetic energy available from the drops.

$$q_2 = M_w \, c_{p_w} \, (t_s - t_\infty - \Delta t_{k_w})$$

where

M_w is the mass of water impinging on the heated area in lb./hr.-ft.^{2}

c_{p_w} is the specific heat of water = 1.0 Btu/lb.-°F

t_{k_w} is the temperature rise due to the kinetic energy of the water droplets

$$t_{k_w} = \frac{U_L^2}{2g\, J\, c_{p_w}}$$

t_∞ is the ambient temperature in °F

The mass of water impinging on the windshield heated area is given by

M_w = 0.225 (E) (Sine Angle 1) (Cosine Angle 2) (U_∞) (LWC)

where

E is collection efficiency

LWC is the liquid water content of the ambient air in gm/m^3

In order to calculate the collection efficiency, Mr. Letton assumed that the aircraft windshield is equivalent to a rectangular half body (semi-infinite rectangle) having a half width the same as the projected height of the windshield heated area. This collection efficiency is expressed as

$$E = \text{Antilog}_e \left[- \frac{\left(\frac{151}{K^2 + 150} + 0.267 + 0.225\, Re_o^{0.28}\right)\left(1.02 - \frac{180}{250 + Re_o}\right)}{(K-0.15)^{0.74}} \right]$$

where

Re_o is the freestream Reynolds number

K is a dimensionless inertia parameter

These parameters are defined as follows

$$Re_o = \frac{2\ a\rho U_L}{\mu}$$

$$K = \frac{2\rho_w\ a^2\ U_L}{9\ \mu\,(\text{Sine angle 1})\ \ H}$$

where

a is droplet radius in feet

H is windshield height in feet

Evaporative Heat Loss (q_3)

The basic equation for the unit heat loss at the windshield surface due to evaporation of some or all of the impinging water droplets is expressed by

$$q_3 = M_{w_e}\ L_s\ F$$

where

M_{w_e} is the maximum rate of evaporation which would occur is lb./hr.-ft.^{2} if the total heated surface area is completely wet.

L_s is the latent heat of evaporation in Btu/lb. of water at the windshield surface temperature, t_s. For this analysis L_s = 1073.46 Btu/lb. since t_s=35°F.

F is the surface wetness factor which varies between 0 and 1. It has a value of 1 for a completely wet surface and a value of 0 for a completely dry surface.

The maximum rate of evaporation, M_{w_e}, may be expressed by

$$M_{w_e} = K_m\ (W_s - W_\infty)$$

where

k_m is the coefficient of mass transfer in lb. dry gas/hr.-ft.^{2}

W_s is the absolute humidity at the liquid-air interface in lb. vapor/lb. dry gas

W is the absolute humidity of the surrounding ambient remote from the liquid-air interface in lb. vapor/lb dry gas

From Daltons law of partial pressures it can be determined that

$$W_s - W_\infty = \frac{.622\ (P_s - P_\infty)}{P_B}$$

where

P is the vapor pressure of saturated water at the ambient temperature, t

P_s is vapor pressure of saturated water at the windshield surface temperature, t_s

P_B is the barometric pressure

The equation for, K_M, the coefficient of mass transfer can be expressed as

$$K_M = \frac{h_{avg}}{0.89\ C_P}$$

Thus the equation for the maximum rate of evaporation, M_{w_e}, can be rewritten as

$$M_{w_e} = 2.91\ h_{avg} \left(\frac{P_s - P_\infty}{P_B} \right)$$

Mr. Letton stated that the maximum rate of heat dissipation due to evaporation will occur when the wetness factor, F, is equal to 1. In his analysis, Mr. Letton assumed that F = 1 as long as the rate of water impingement on the windshield, M_w, is greater than the rate of evaporation M_{we}. For cases where the rate of impingement is less than the maximum rate of evaporation which could occur, he then assumed the $F = M_w/M_{w_e}$.

Heat Loss Due to Radiation (q_4)

The basic equation for the unit heat loss at the windshield outer surface due to radiation is expressed by

$$q_4 = \sigma \epsilon (T_s^4 - T_\infty^4) F_r$$

where

σ is the Stefan-Boltzmann constant which equals 0.1713×10^{-8}

ϵ is the emissivity of the windshield, which is assumed to be glass and have a value of 0.94

T_∞ is the absolute temperature of the ambient air in °R

F_r is the radiation shape factor, which is assumed to have a value of 1 for this analysis

In writing this computer program, Mr. Letton made several major assumptions including the following:

(1) The local velocity over the windshield was assumed to be equal to the freestream or aircraft velocity.

(2) Calculation of collection efficiency is based on the assumption that an aircraft windshield is equivalent to a rectangular half body.

(3) The wetness factor, F, is equal to 1 as long as the rate of water impingment (M_w) is greater than the rate of evaporation (M_{w_e}).

Mr. Letton felt that the above assumptions were conservative.

RESULTS

Over the past several months we have made numerous computer simulations using this program in order to evaluate it. After reviewing these cases, we can make the following general observations:

1. The windshield heat requirement is decreased by increasing windshield height.

2. The windshield heat requirements decrease as the distance to

the stagnation point increases.

3. Increasing the windshield installation angle generally decreases the heat requirement.

4. Generally, for the same altitude and ambient temperature, heat requirements during intermittent maximum icing conditions are more severe than for continuous maximum icing conditions.

These observations are illustrated by the results presented in Figures 6 through 11. These figures show windshield heat requirements versus the distance from the windshield to the aircraft stagnation point for windshield heights of 1.5, 2.5, and 3.5 feet. Figures 6 through 8 present this data for intermittent maximum icing conditions at windshield installation angles of 20°, 40°, and 60° respectively. Similarly, data for continuous maximum icing conditions are presented in Figures 9 through 11. In order to obtain the comparison of the results shown in these figures, the following parameters were held constant: altitude = 8000 ft. median droplet diameter = 30 microns, outside ambient temperature = -22°F and aircraft speed = 200 knots. It should also be noted that the maximum heat requirement does not always occur at the same altitude and median droplet diameter.

The generalities presented above hold true for the majority of the configurations which have been investigated, however, exceptions do exist. This fact, again, illustrated the need for individual heat requirements analysis for each aircraft configuration requiring anti-icing capabilities.

Mr. Letton theorized that the reason that the heat required decreased as the windshield height increased was because collection efficiency, E, decreases with increasing height. Increasing the distance from the stagnation point decreases the required heat because of a reduction of the convective heat transfer coefficient with increasing distance from the stagnation point. Increasing the angle has the same result of decreasing the collection efficiency as increasing the windshield height.

In order to show the promise of this program or one similar to it, the height, installation angle, and the distance to the stagnation point of a typical small aircraft windshield were input to the program for both the intermittent maximum and continuous maximum icing conditions. Some selected results are presented in Figure 12 for the intermittent maximum icing conditions, and in Figure 13 for the continuous maximum icing condition at different ambient temperatures.

After reviewing icing condition documentation and the theory behind calculating the heat requirements necessary to maintain an anti-ice windshield condition, the following recommendations are made:

1. Each aircraft windshield should be thoroughly analyzed to determine the most efficient heat requirement necessary to maintain an anti-ice condition.

2. A continuing effort should be maintained to obtain actual aircraft icing data (both test and actual occurance) necessary to evaluate existing computer programs.

3. A continuing effort should be maintained by both government and industry to develop computer programs for simulation of all possible icing conditions.

4. A study should be made to determine the effect of helicopter rotor turbulence on such parameters as droplet diameter and catch efficiency.

5. The results of Mr. Lettons program are probably conservative due to the assumptions he was required to make.

PPG Industries is very much interested in the research programs listed above and will assist in the solution of these heat requirements problems in any way possible.

PPG HEATING CAPABILITIES FOR AIRCRAFT TRANSPARENCIES

Once the heat requirement for a particular aircraft has been determined, regardless of the method used, PPG Industries has a wide range of coatings available for providing de-icing, anti-icing, or defogging capabilities for helicopter transparencies. First among PPG products is our NESA® heating film which has proven over many years to be an extremely dependable coating. In recent years, PPG has introduced NESATRON® and AIRCON® which together with NESA gives PPG the total capability for providing de-ice, anti-ice, and defog transparencies for all aircraft regardless of the voltage system employed. The following is a review of each of these systems together with its capabilities and its present use in the helicopter field.

AIRCON®

PPG's Aircon is a system of extremely fine wires embedded in the interlayer adjacent to the outboard glass ply. The wires most

commonly used are between .4 and .8 mils in diameter and are typically sewn into the interlayer at a .028" spacing. The diameter and type of wire used and the spacing between wires is dependent upon the size of the heated area and the final resistance desired. With this system, total resistances in the range of one (1) ohm are produced, thus, this heating system is ideal for use on aircraft with limited power available. (Approximately 28 volts). At present, windshields with power dissipation as high as 3 watts/in.^{2} are being produced without sacrificing good optical quality.

PPG is continuing to improve the Aircon system through development efforts at our manufacturing and research facility. These efforts include the use of wires of varying diameters and metallic content, as well as experimentations with Aircon mats tied in series so as to heat large areas.

In the helicopter field, PPG has produced approximately nine ship sets of Bell UH-1H heated windshields employing Aircon as a means of de-icing. Approximately 343 square inches across the center of the panel is heated with a constant power density of 2.6 watts/in.^{2}. One of these transparencies was tested by the U.S. Army System Test Activity, Edwards Air Force Base, California, with the following conclusions as reported in USAASTA Project #73-04-4.

a) "The heated portion of the heated glass windshield provides satisfactory anti-ice/de-ice capability."

b) "The optical characteristics of the heated glass UH-1H windshield in a non-icing environment were satisfactory during flight in daylight, twilight, and night conditions."

c) "The addition of electrical power to the heated section of the windshield did not change the optical characteristics of the windshield."

One shortcoming was noted during the evaluation with the recommendation being:

> "further evaluation should be conducted to determine the optimum size, shape, and location of the heated portion of the windshield for the UH-1H helicopter".

PPG is continuing to work with Bell Helicopter and others on the UH-1 windshields in an effort to produce the most effective transparency for the mission profile. While the UH-1 is the only helicopter presently employing the Aircon system, there are many

private industry aircraft such as the Beech King Air which totally rely on Aircon for de-icing.

NESA®

PPG's Nesa coating is now used on a wide range of products. Presently, Nesa is used to provide de-icing and de-fogging capability on Boeing Vertol's CH-46, CH-47, the Sikorsky CH-53 and well over 100 other commercial and military aircraft. This tin oxide coating is applied to the inboard surface of the outboard glass ply for de-icing. De-fogging on the CH-46, CH-47, and the CH-53 is accomplished through the use of .090" thermally tempered glass with a thin interlayer which allows sufficient heat to be transmitted to the inboard glass ply to maintain this surface above the dew point. Application occurs with the glass temperature in the range of 1100°F, thereby producing a very hard and durable coating. Resistivities as low as 15 ohms per square are presently being produced with a great degree of uniformity in heat output over the entire heated area. Generally, light transmission in excess of 70% can be achieved on thin glass with a very low resistivity.

Nesa is by far the most commonly used heating film today because its' resistivity range is ideal for helicopters and other aircraft utilizing power systems ranging from 195 to 400 volts.

NESATRON®

Nesatron is a relatively new coating developed by PPG's research and development facility in Harmarville, Pa. At present, Nesatron can be applied with surface resistivities as low as 5 ohms per square with light transmission in the neighborhood of 75%. This coating is an indium oxide film applied by sputtering a water cooled cathode of indium metal alloyed with some tin, in an atmosphere consisting predominately of argon and oxygen. The sputtering occurs well below the softening point of glass thus application of this coating can be applied to almost any glass surface without affecting contour or temper of the glass ply. A primary advantage of this film is the sputtering process which allows for a great degree of thickness control. Coatings can therefore be produced which require a large degree of grading from one bus bar to another or across the width of the panel.

Nesatron has most recently been used to coat the pilot and copilot windshields for the Sikorsky UTTAS. Final testing of the completed windshields has shown that the coating is in fact, very uniform over the entire surface.

CONCLUSION

PPG Industries is well aware of the many considerations necessary during the design of an electrically heated transparency. Items such as size, configuration, angle of installation, and windshield height are significant factors in determination of the heat dis-ipation required. Development of a computer program for heat requirement analysis will be a significant step forward in aircraft transparency design. PPG is continuing its research into the program with the hope that sufficient data and experience can be gathered to up-date the program to a state where it reasonably predicts heat requirements for a particular aircraft.

In addition to this endeavor, the conductive films produced by PPG; Nesa, Nesatron, and Aircon are continually being upgraded to provide the most uniform coating possible.

References

1. Bowden, D. F.; Gensemer, A. E.; Skeen, C. A.; Engineering Summary of Airframe Icing Technical Data. Federal Aviation Agency Report ADS-4, 1964

2. Letton, G. C.; An Analytical Investigation of Aircraft Windshield Anti-Icing Systems. Masters Thesis, Ohio State University, 1972.

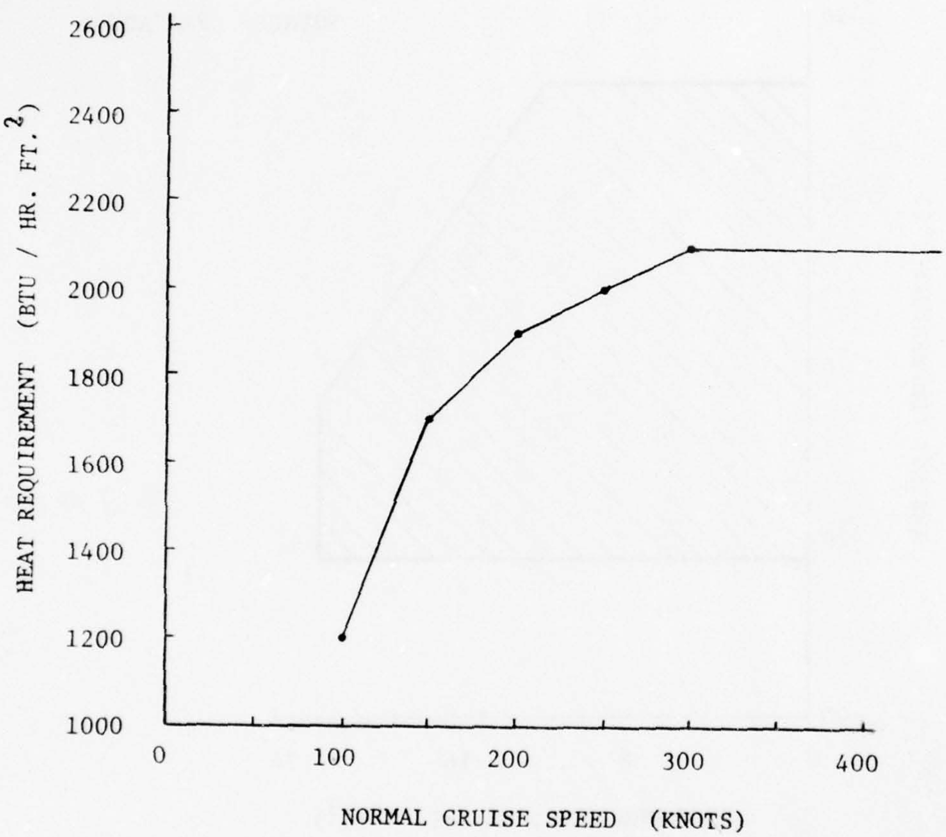

FIGURE 1. WINDSHIELD HEAT REQUIREMENTS AS DEFINED BY MIL-T-5842A

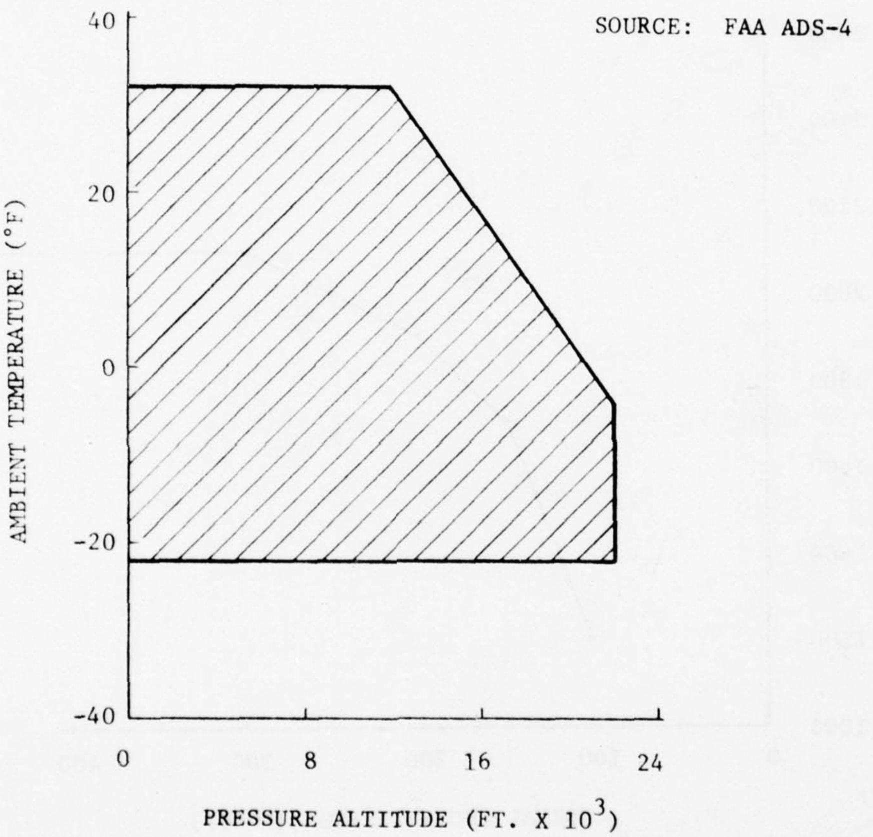

FIGURE 2. CONTINUOUS MAXIMUM ATMOSPHERIC ICING CONDITIONS - STRATIFORM CLOUDS

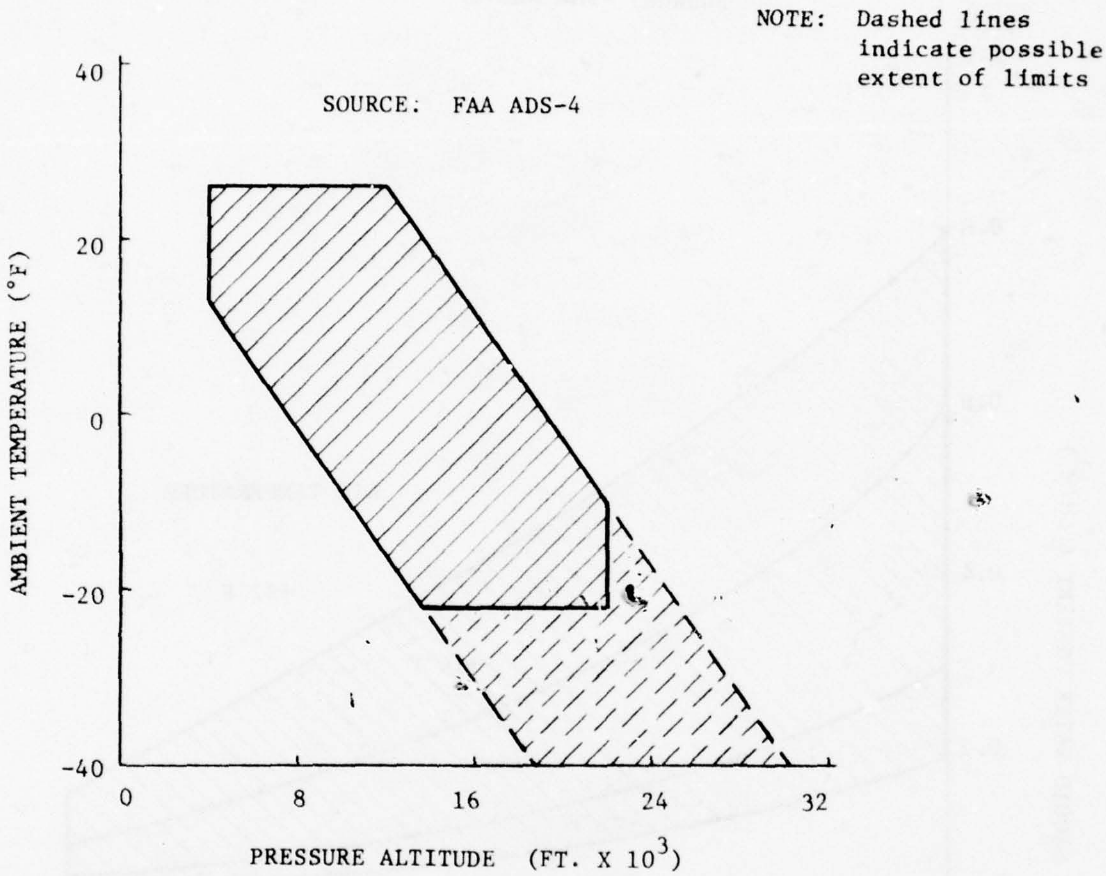

FIGURE 3. INTERMITTENT MAXIMUM ATMOSPHERIC ICING CONDITIONS - CUMULIFORM FORMS

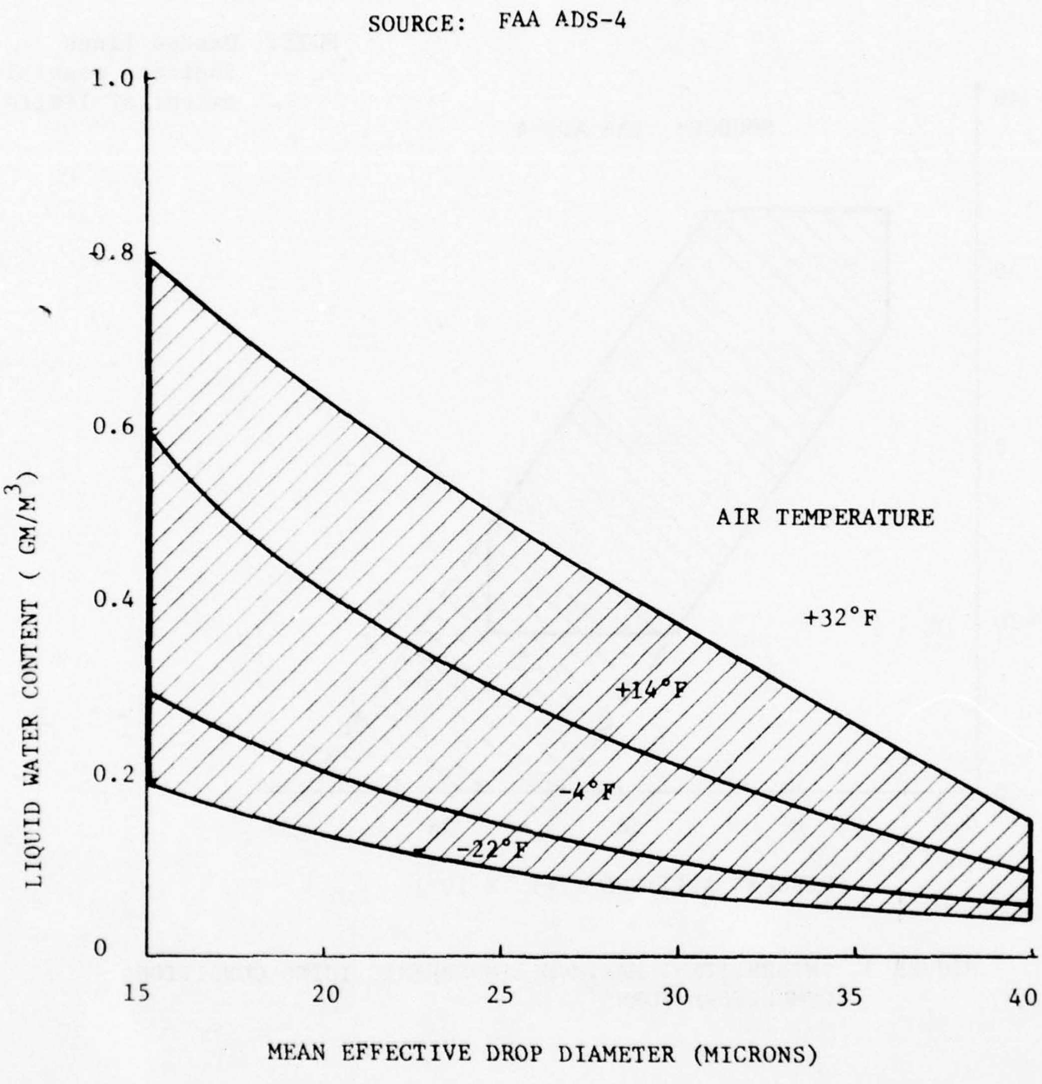

FIGURE 4. CONTINUOUS MAXIMUM ATMOSPHERIC ICING CONDITIONS - STRATIFORM CLOUDS.

LWC VERSUS MEAN EFFECTIVE DROP DIAMETER

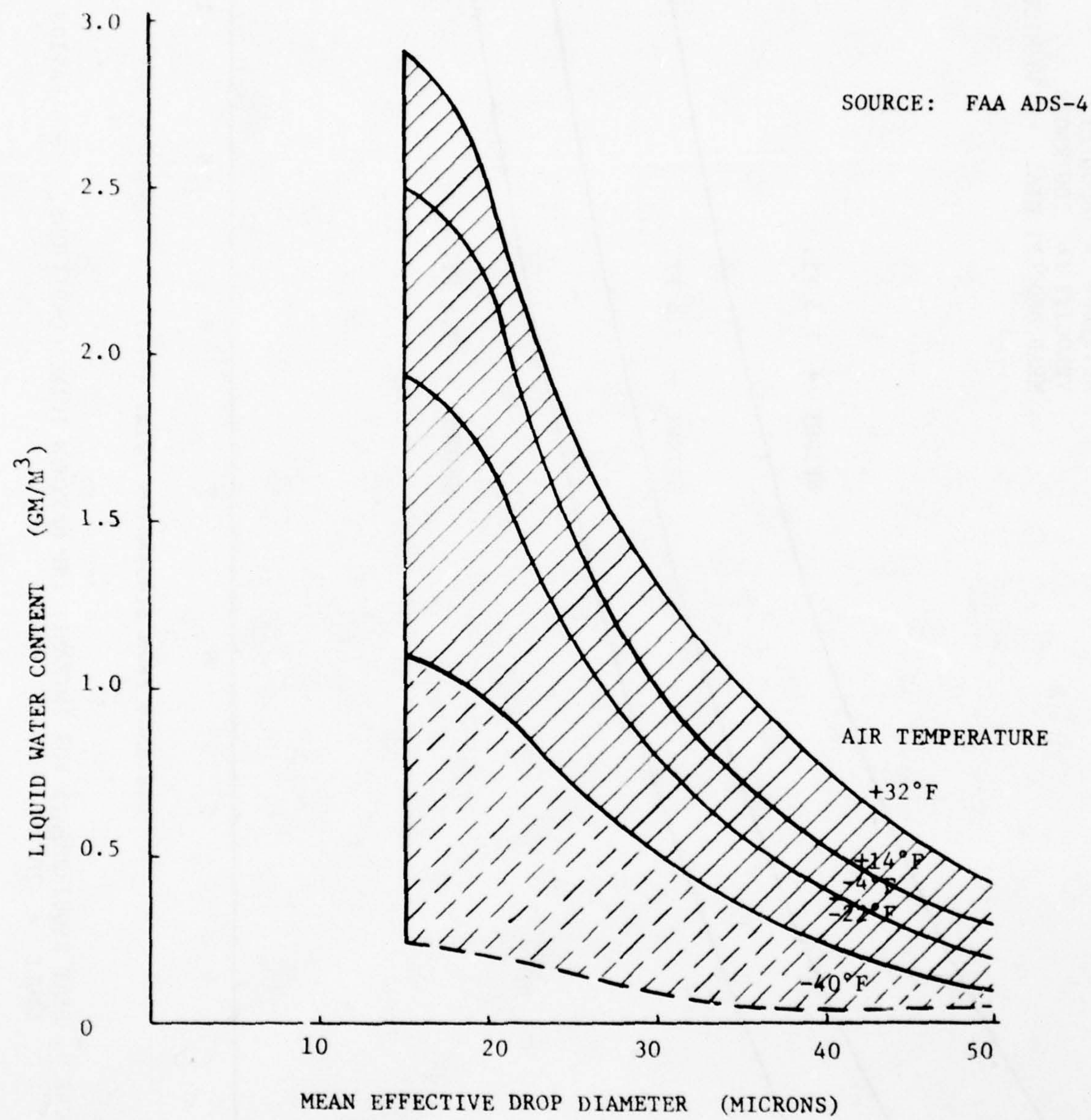

FIGURE 5. INTERMITTENT MAXIMUM ATMOSPHERIC ICING CONDITIONS - CUMULIFORM CLOUDS.

LWC VERSUS MEAN EFFECTIVE DROP DIAMETER

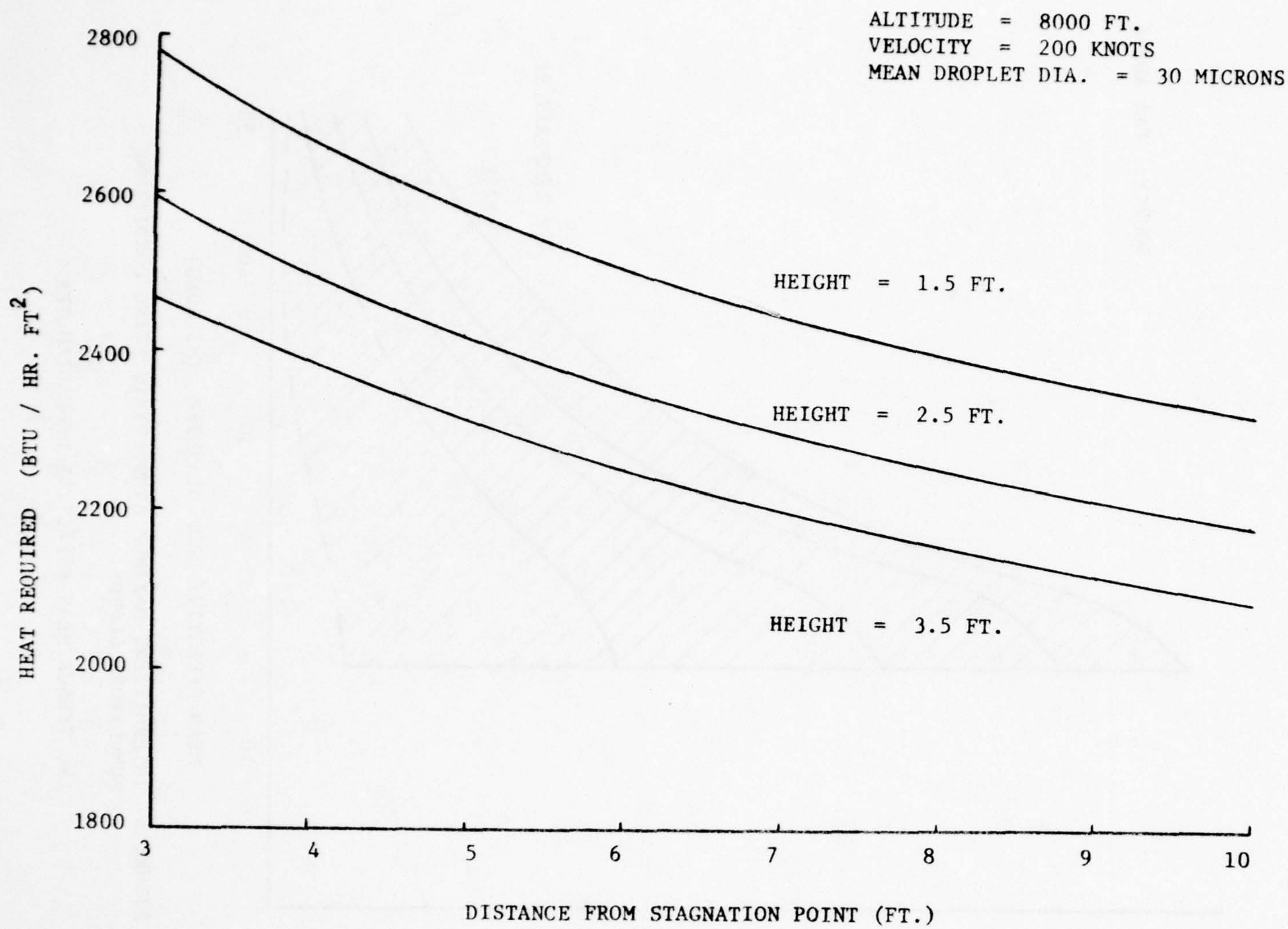

FIGURE 6. HEAT REQUIREMENTS FOR INTERMITTENT MAXIMUM ICING CONDITIONS, INSTALLATION ANGLE = 20°

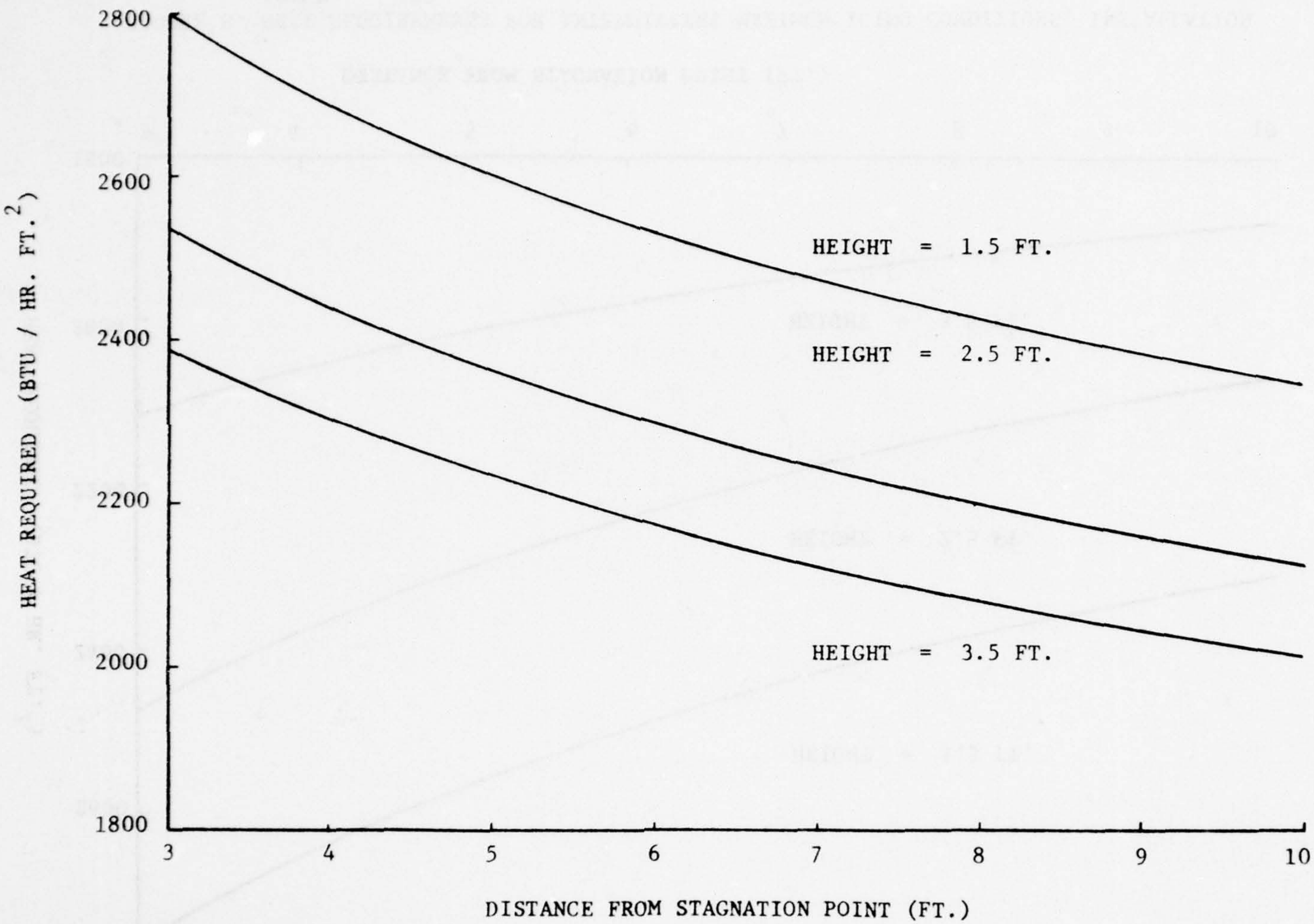

FIGURE 7. HEAT REQUIREMENTS FOR INTERMITTENT MAXIMUM ICING CONDITIONS, INSTALLATION ANGLE = 40°

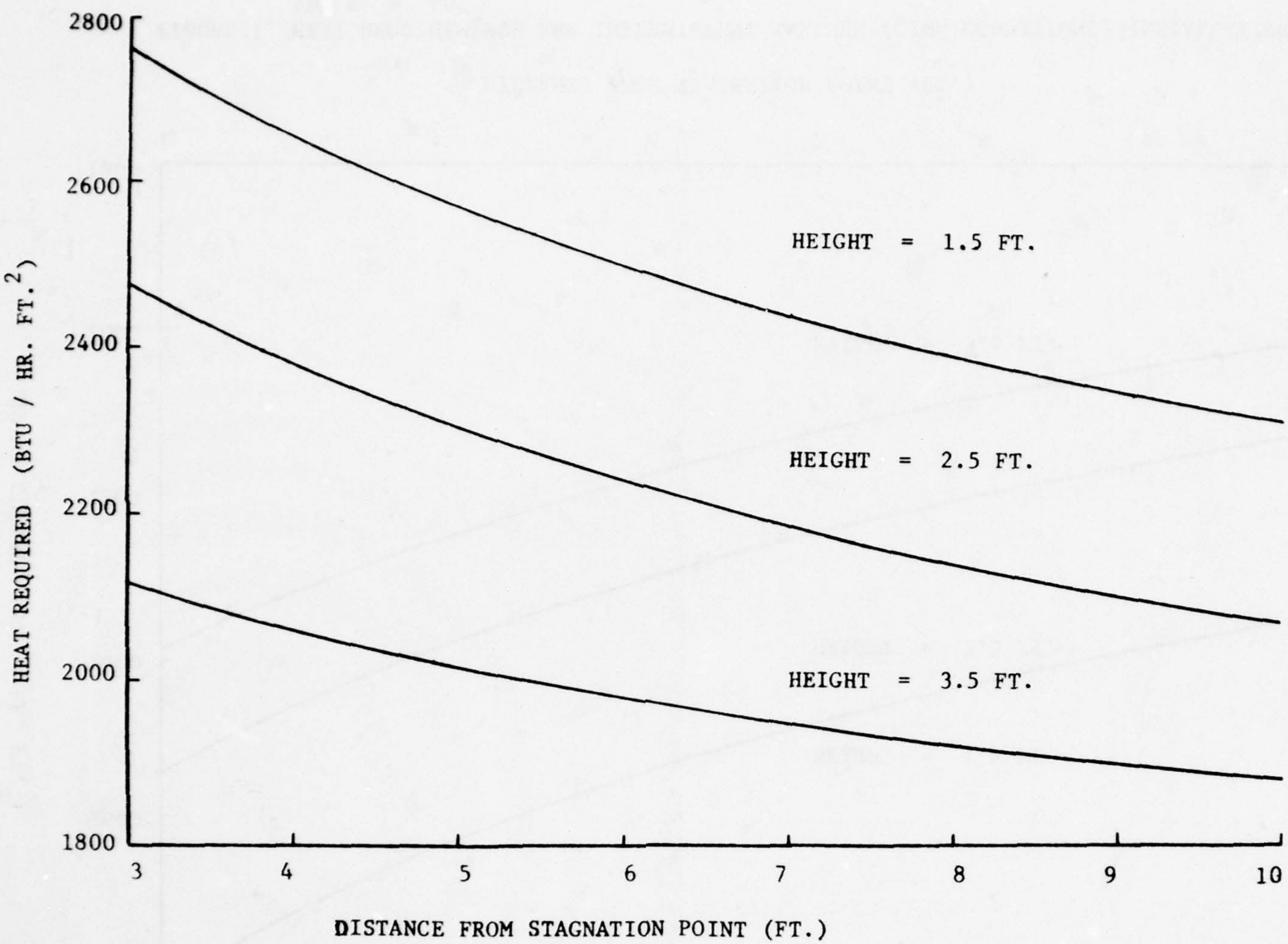

FIGURE 8. HEAT REQUIREMENTS FOR INTERMITTENT MAXIMUM ICING CONDITIONS, INSTALLATION ANGLE = 60°

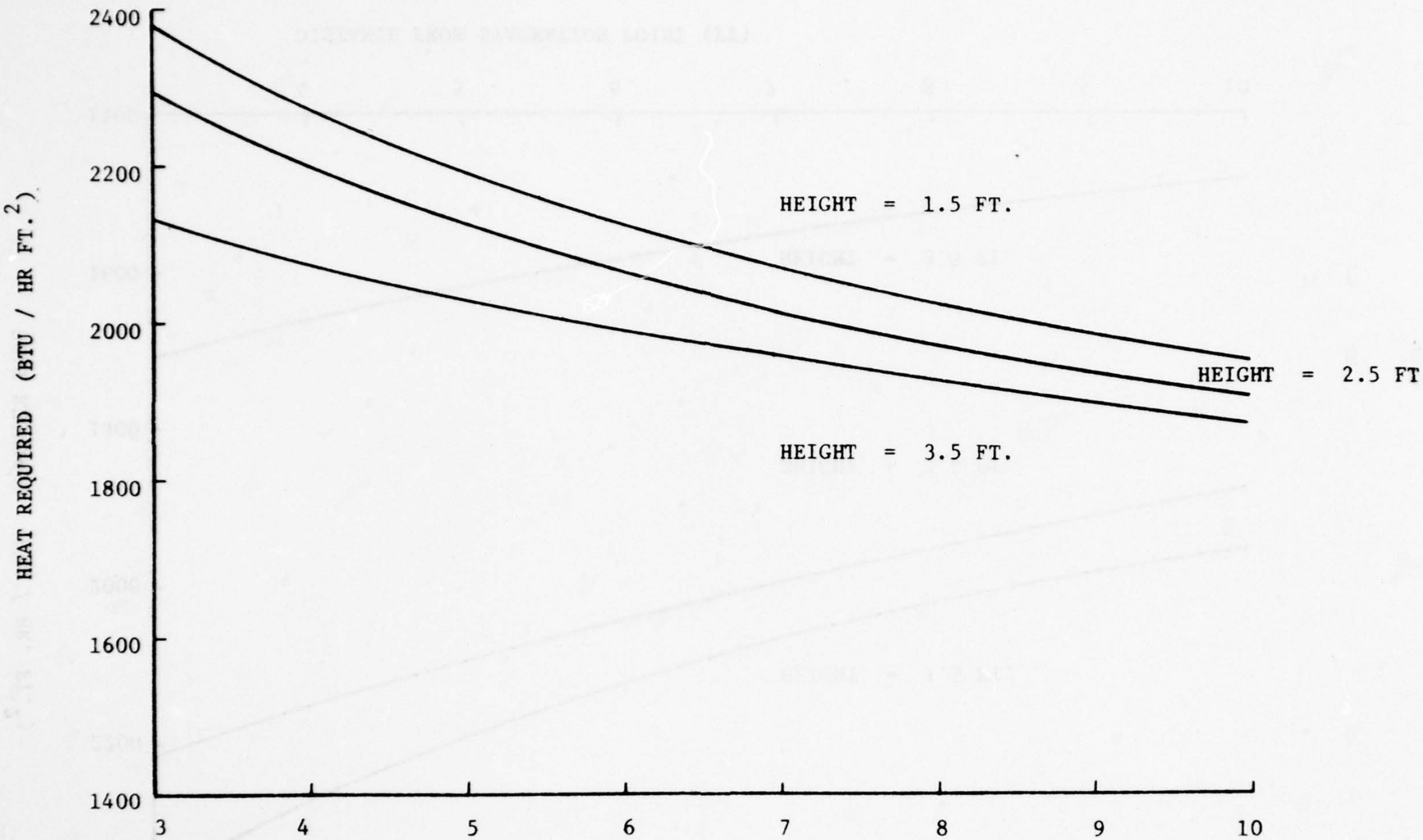

FIGURE 9. HEAT REQUIREMENTS FOR CONTINUOUS MAXIMUM ICING CONDITIONS, INSTALLATION ANGLE = 20°

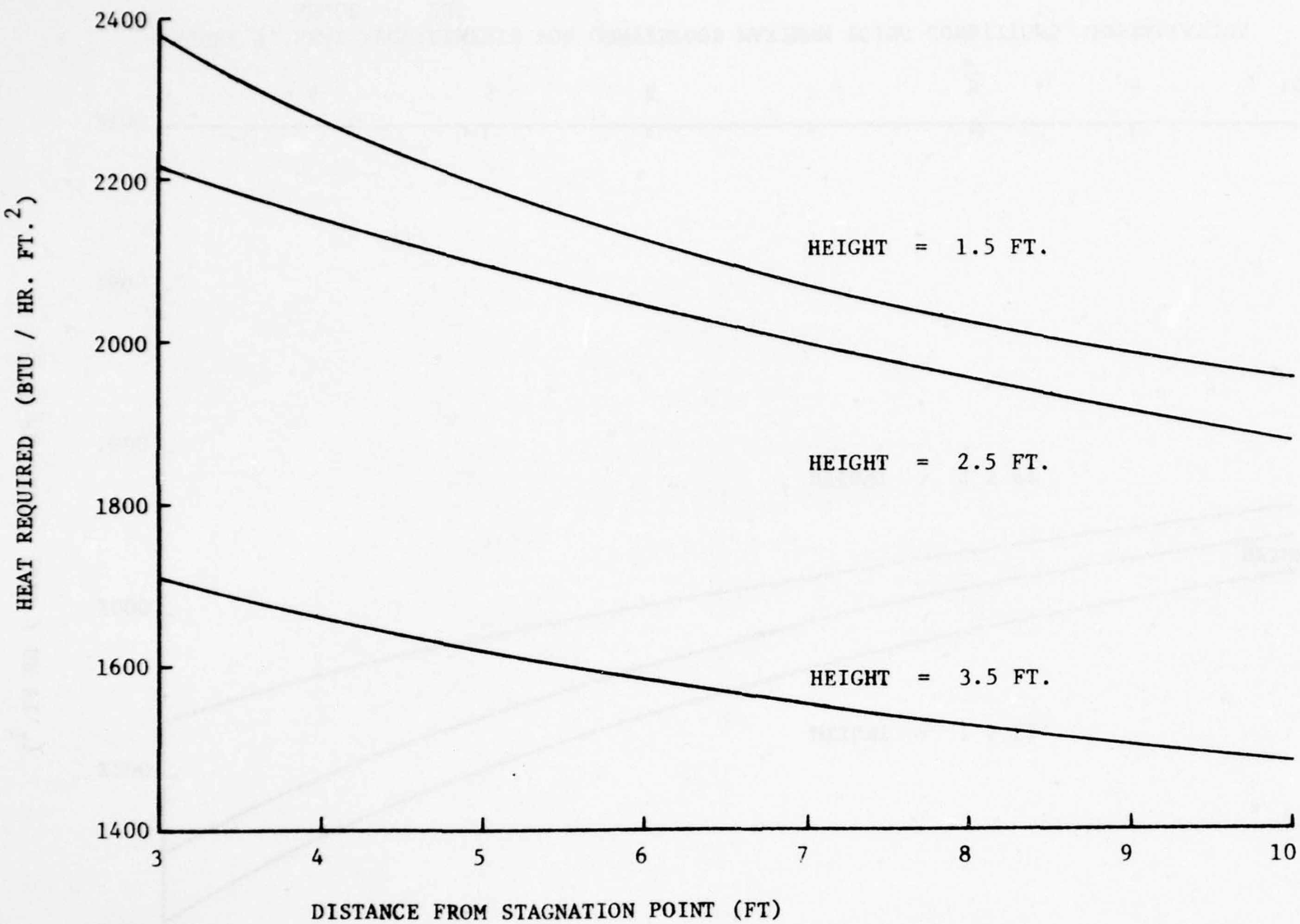

FIGURE 10. HEAT REQUIREMENTS FOR CONTINUOUS MAXIMUM ICING CONDITIONS, INSTALLATION ANGLE = 40°

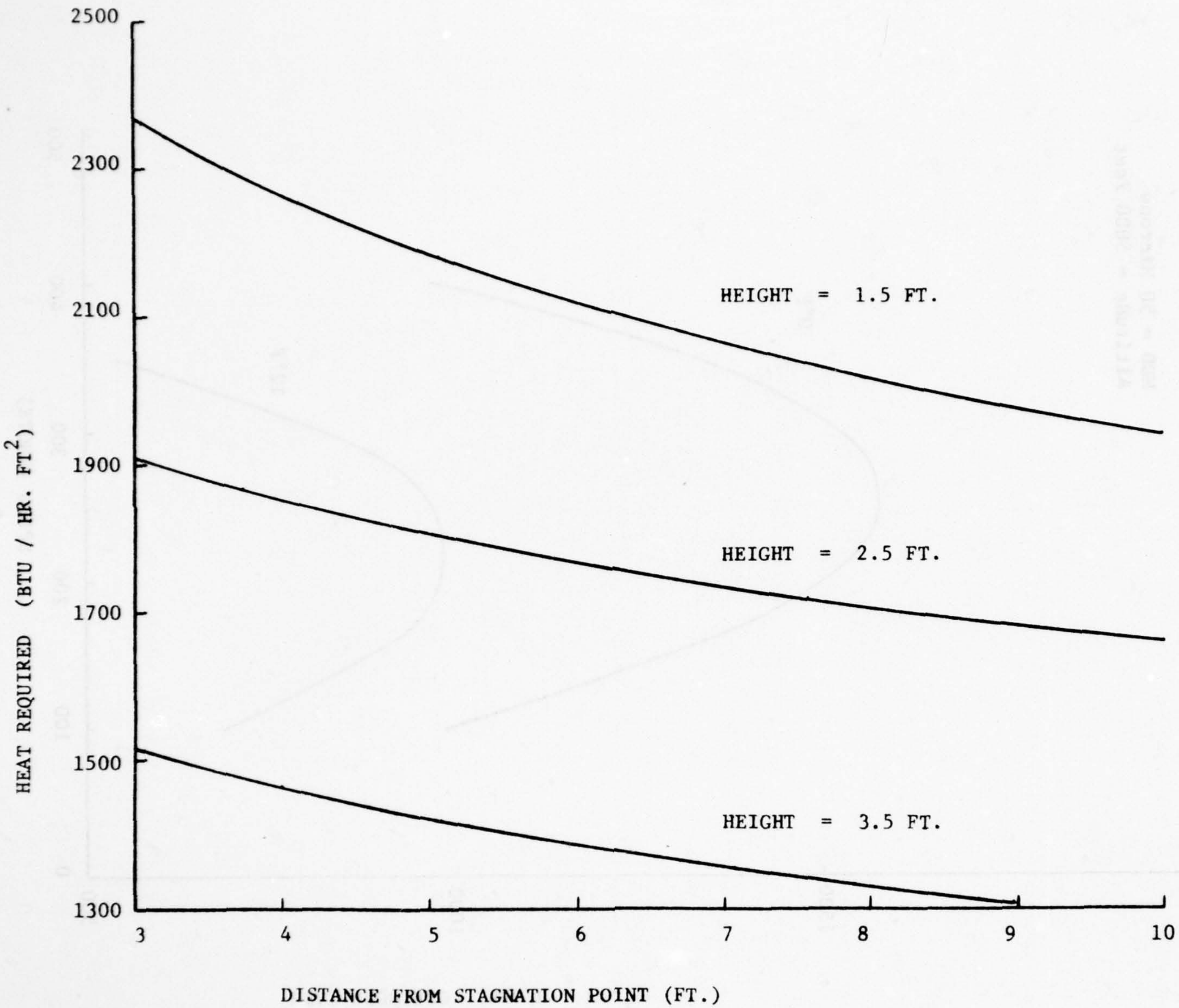

FIGURE 11. HEAT REQUIREMENTS FOR CONTINUOUS MAXIMUM ICING CONDITIONS, INSTALLATION ANGLE = 60°

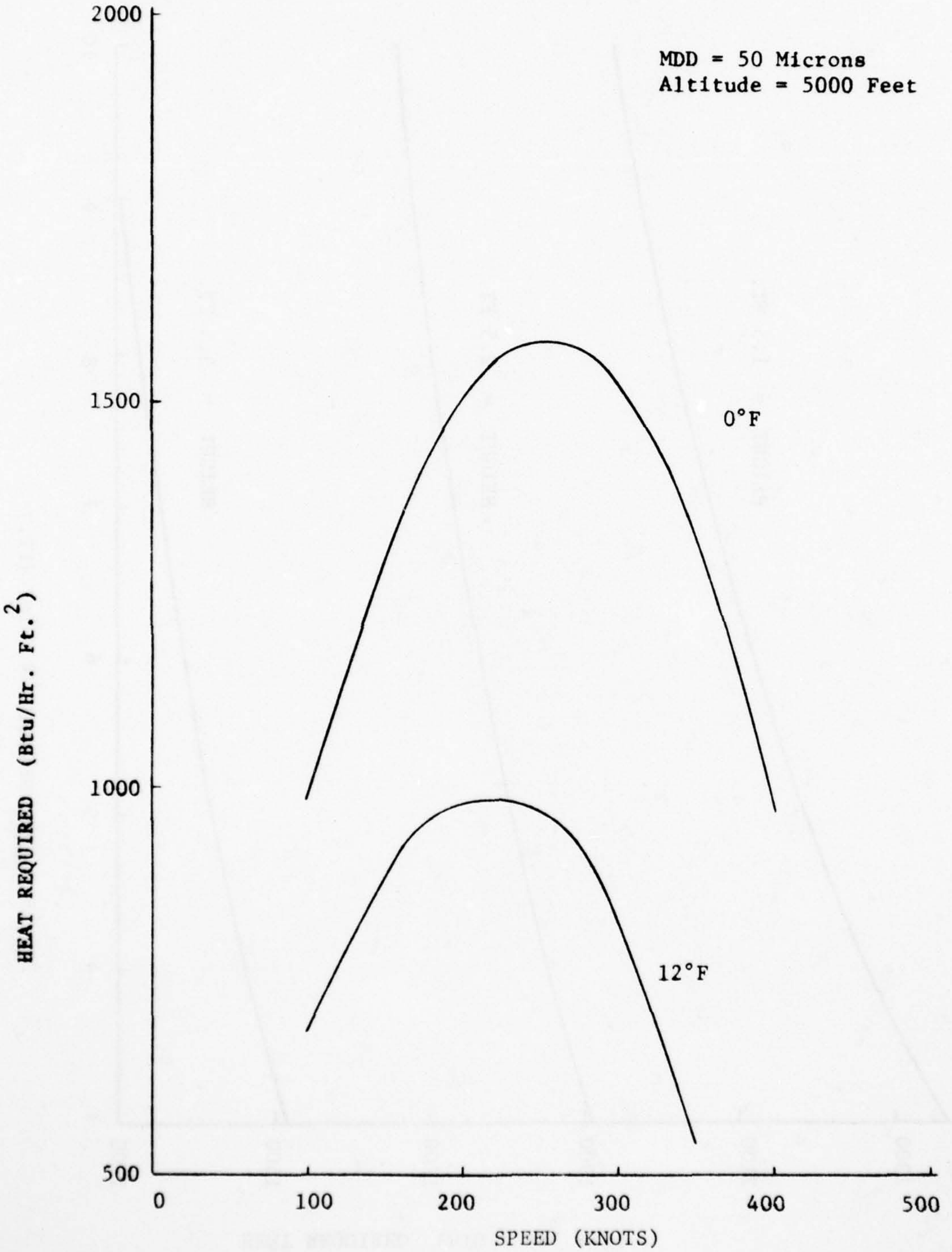

FIGURE 12. INTERMITTENT MAXIMUM HEAT REQUIREMENTS AT 5000 FEET

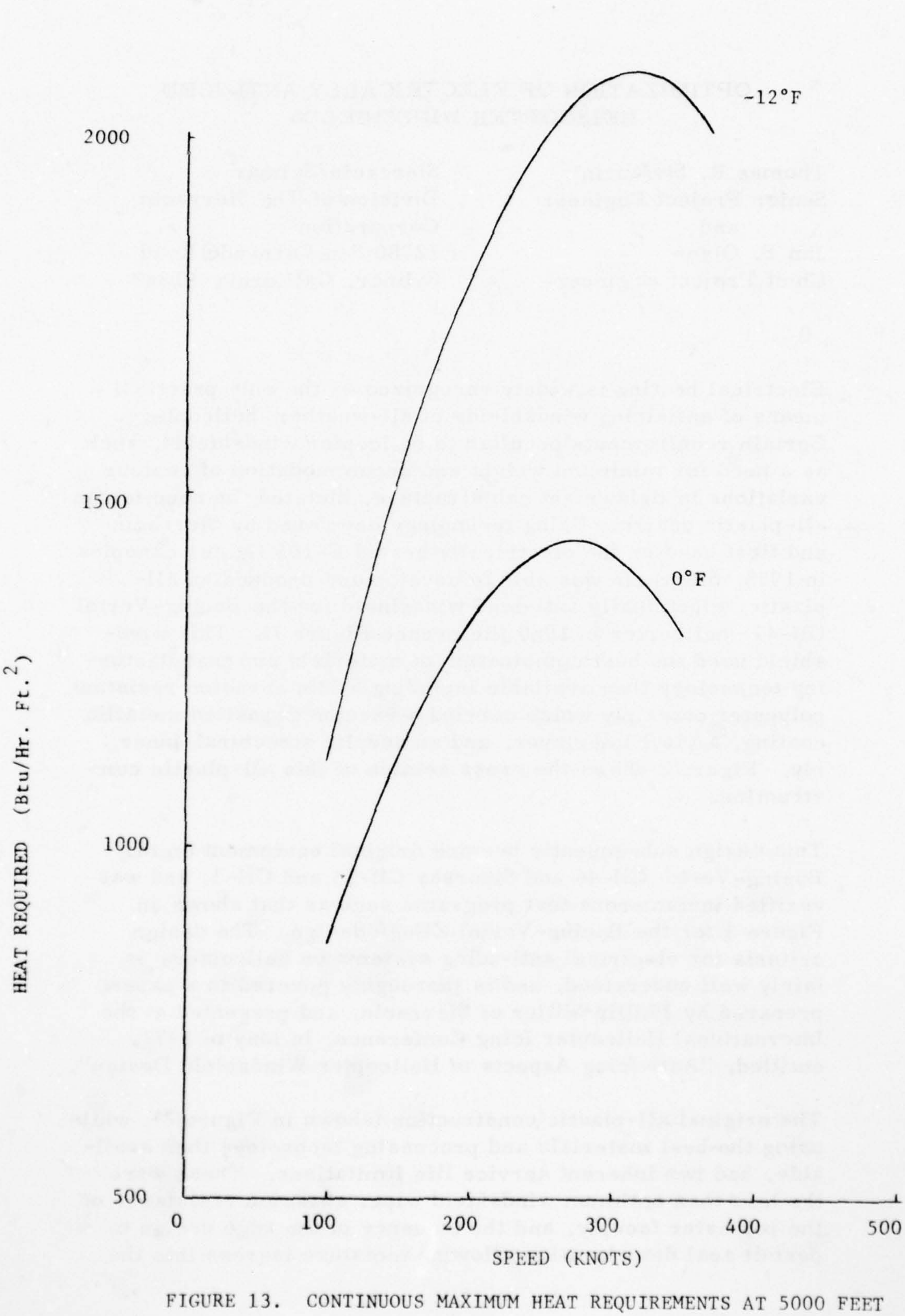

FIGURE 13. CONTINUOUS MAXIMUM HEAT REQUIREMENTS AT 5000 FEET

OPTIMIZATION OF ELECTRICALLY ANTI-ICED HELICOPTER WINDSHIELDS

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Electrical heating is widely recognized as the only practical means of anti-icing windshields of all-weather helicopters. Certain requirements peculiar to helicopter windshields, such as a need for minimum weight and accommodation of contour variations in lightweight cab structure, dictated the need for an all-plastic design. Using technology developed by Sierracin and first used on the electrically heated F-106 fighter canopies in 1958, Sierracin was able to develop and produce an all-plastic, electrically anti-iced windshield for the Boeing-Vertol CH-47 helicopter in 1960 (Reference Figure 1). This windshield used the best combination of materials and manufacturing technology then available including a thin abrasion resistant polyester outer ply which carried a vacuum deposited metallic coating, a vinyl interlayer, and an acrylic structural inner ply. Figure 2 shows the cross section of this all-plastic construction.

This design subsequently became original equipment on the Boeing-Vertol CH-46 and Sikorsky CH-53 and CH-3, and was verified in numerous test programs such as that shown in Figure 3 for the Boeing-Vertol CH-46 design. The design criteria for electrical anti-icing systems on helicopters is fairly well understood, and is thoroughly covered in a paper prepared by Phillip Miller of Sierracin, and presented at the International Helicopter Icing Conference, in May of 1972, entitled, "Anti-Icing Aspects of Helicopter Windshield Design".

The original all-plastic construction (shown in Figure 2), while using the best materials and processing technology then available, had two inherent service life limitations. These were the less than optimum windshield wiper abrasion resistance of the polyester faceply, and the tendency of the edge design to permit seal deterioration allowing moisture ingress into the

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interlayer and electrical coating area. This moisture sometimes led to delamination and/or electrical failure near the edges. The performance of this construction was nonetheless quite adequate until the notoriously abrasive dust and the very warm, humid climate of Vietnam became the normal operating environment for many military helicopters.

These conditions, coupled with a timely breakthrough by Corning Glass Works in thin glass chemical strengthening technology, caused Sierracin to embark on a high priority program to provide a glass faced version of essentially the same proven windshield. Due to the urgent military need for an improved windshield, very little innovation was possible in the new design, except that thin glass replaced the polyester plastic outer ply. Figure 4 shows the all-plastic and early composite side-by-side for comparison. However, the original Corning chemically strengthened glass was not suitable for combat helicopters in that it was easily broken by small stones drawn up by the rotor wash during hovering, and when it broke, residual visibility was very limited. Sierracin and Corning worked cooperatively to perfect a specialized version of this glass which had greater resistance to impact damage coupled with a large break pattern that permitted pilot vision even after breakage.

Owing to the large difference in the coefficients of expansion of glass and plastic, it was necessary to develop a new laminating technology that would permit us to join these two materials without incurring high thermally-induced stresses which would cause contour variation as well as a potential service life limitation. This new glass and plastic composite windshield construction was very successful in extending service life, and in fact, very few service failures with this design have occurred. However, because it was executed almost literally in the heat of battle as a near-term fix, this first generation glass/plastic design was not optimum regarding cost, weight, or service life.

Not content with this interim product, Sierracin decided to embark on a company-funded program to bring the helicopter windshield to its full potential. Rather than merely further

modifying the original all-plastic design, the board was wiped clean and we embarked on a program with no preconceived notions about the construction of the windshield other than that the outer facing must be glass because of the abrasion problem with plastics. All of the helicopter manufacturers as well as the using commands were consulted, and a list of performance criteria drawn up and prioritized. These boiled down to four major categories; SERVICE LIFE, SAFETY, WEIGHT and COST.

One major criterion was service life, which dictated the use of the glass outer ply, but said very little about the interlayer or inner ply or edge attachment constructions, leaving a number of possibilities in these areas open to us.

The other prime criterion was safety, and here we discovered a serious omission on the specifications for the various helicopter windshields. None of them address the common sense requirement that a windshield remain in one piece after penetration by a bullet or other object. If it does not, the airstream (up to 225 knots in modern military helicopters) could cave in the broken windshield, perhaps injuring the crew, certainly making the control of the aircraft tricky and possibly even causing major structural damage to the fuselage from the great blast of air that would be scooped in. Let me repeat; there were no written requirements for the windshield of a combat helicopter to withstand wind-loading after ballistic penetration, which event must be considered a probable failure mode as indicated by the fact that all military helicopter windshield specifications reviewed contain a ballistic spall test requirement. Fail-safe design is predicated on non-catastrophic reaction to all likely failure modes. After all, what is the advantage in providing a windshield which has sufficient spall resistance to prevent showering the crew with flying fragments, but which will subsequently collapse and wrap itself around their heads. We, therefore, imposed this requirement upon ourselves, feeling it was within the state-of-the-art and could not in good conscience be ignored. The other prime safety consideration is that adequate visibility should be retained in all conditions, even in the event of glass breakage.

Next in order, the windshield should have minimum weight and minimum manufacturing costs consistent with attainment of the primary requirements.

As previously mentioned, a specialized type of chemically strengthened glass was developed as the optimum for helicopter usage, and indeed, the experience on the interim CH-47 glass/plastic windshield had borne out this selection. Therefore, we settled on this proven glass for the outer ply of any design under consideration. Analysis showed that .085 inch thick glass by itself was as stiff as the original all-plastic laminated windshields, so that any inner ply added to this could be considered strictly as a fail-safe member, and as a shield to protect the heating element.

The next major facet of our development program was to select the optimum inner ply. The candidate materials included:

- MIL-P-8184 as-cast acrylic
- MIL-P-25690 stretched acrylic
- Rohm & Haas Plex 70 (a tougher as-cast acrylic)
- Polycarbonate
- Glass

Six windshields were fabricated for the purpose of evaluating each inner ply material listed here, with the exception of Plex 70 which was eliminated because of its milky appearance at elevated temperatures. Table 1 shows the cross sections of these six test windshields. You will notice, the faceply is the same on all specimen windshields, .085-inch thick Chemcor glass, except for Specimen No. 1, the all-plastic design which incorporated the .062 inch thick polyester plastic. The inner ply was varied as shown, including Number 4, which has a glass inner ply. The interlayer was the same on all the test windshields shown. Specimen No. 2 with the 1/8-inch acrylic inner ply is the base line representing the non-optimized CH-47 glass/plastic design. The thickness of the plastic inner plies were selected for each candidate material based on fail-safe considerations, and glass was included because it offers economy of manufacture, in exchange for toughness and light weight. These thicknesses are specific to the CH-46 side

windshields, and should be reconsidered on each new application to reflect the size, curvature and wind-load conditions of the aircraft in question.

The six specimen windshields shown in Table 1 were penetrated with two 30 caliber ball projectiles (Reference Figure 5), evaluated for spall and visibility and then subjected to a wind-loading of 1.6 psi minimum for fail-safe testing. All windshields passed except the all-glass, which failed to withstand wind-loading after ballistic penetration as shown in Figure 7. The windshield would have "blown" out of the frame except that the air bladder used to simulate wind-loading limited the excursion. Also, the residual visibility was considerably better on all of the glass/plastic windshields than on the all-glass part shown here. Figure 6 shows the basic test set-up for the fail-safe test and Figures 8 through 13 show the tested windshield and its applicable spall witness, which was aluminum foil over one inch thick styrene foam boards.

The Matrix, Table 2, is a configuration summary sheet for the CH-46 which compares weight, ballistic spall test evaluation and fail-safe wind-load test evaluation for all cross sections considered. Cross section B, the early composite CH-47 design used as the base line weighs 16.8 pounds per panel. Compared to this base line, cross section E and cross section F, which became our two favored designs, show a weight savings of 12 and 20 percent respectively; that is, 4 and 7 pounds per shipset, while all-glass cross section D shows an increase of 20 percent, or 7 pounds per shipset. You will notice that all configurations passed the spall test with polycarbonate showing superior performance. In the fail-safe wind-loading, however, all passed except the all-glass configuration.

Our final selection was .080-inch thick stretched acrylic based on these ballistic and fail-safe considerations, coupled with this material's excellent recognized service performance in many applications including the earlier glass/plastic CH-47 design. Polycarbonate might have been selected based on its superiority in weight and safety aspects, but uncertainty at that time about the performance of coatings required to

protect this material against solvent attack and scratching prevented, or at least postponed, its selection.

The next major consideration was optimizing the edge attachment for the composite design. The original edge attachment was primarily a fabric reinforced plastic shim, included to increase the bearing strength of the edge and bring the edge up to flushness with the faceply. It was not considered to contribute to seal protection or faceply retention, but solely as a strengthening filler. Experience on that design indicated the need to protect the seal, so modification was incorporated on the interim glass/plastic CH-47 which essentially took these same materials and laminated up an edge attachment that did extend over the glass faceply. It was, however, expensive from both materials and labor standpoints, and it was felt that considerable improvement could be made in this area. Figure 4 shows both of these configurations.

Because the cross sections are usually constant around the perimeter of a windshield, the most attractive solution was to employ an extrusion with more aerodynamically smooth overlap, and one that could contain a detail to establish glue-line thickness. Figure 14 shows three types of extruded edge attachments, two in production and the third a proposed design. It was also a consideration that this edge attachment should be structurally bonded to the outer ply of the glass which in the case of this optimum composite configuration, is the prime load carrying member. One of the failings of the original design is that any out-of-contour condition would force the inner ply away from the faceply creating a peeling action at the expense of seal integrity and in some cases, could cause delamination at the edge. These considerations prompted a broad material search for both an appropriate extrudable edge attachment material and a suitable adhesive sealant system to be tolerant of differential expansion effects and provide seal integrity under all conditions. The resultant system consisted of a structural thermoplastic which had already proven itself in terms of resistance to the outdoor military environment in other uses, and an excellent polysulfide adhesive-sealant.

The next major aspect of the program once the glass faced plastic design was selected, was to provide a more economical means of joining these materials, and hopefully, one which would even further reduce the residual stress due to the joining process. Two programs were conducted in parallel, one of which was aimed at developing new technology to be used with the tried and proven polyvinyl butyral or PVB interlayer, and another based on the use of cast-in-place technology, using castable materials such as silicones and polyurethanes.

A major breakthrough in PVB laminating, called LoLam, was accomplished on this program. Because this is proprietary, I cannot go into details except to say that the net effect is improved service life and contour control because the residual stress is reduced to near zero at room temperature. That is coupled with a reduced manufacturing cost compared with the method developed for the original CH-47 composite.

Tests to evaluate candidate interlayer materials were: Accelerated weathering (humidity, elevated temperature and UV), bond tensile, bond shear, bond peel, tensile creep, and thermal shock, to name a few.

The PVB interlayer using the LoLam process was selected based on its outstanding performance in these tests, its known performance in years of actual service, and its apparent manufacturing and tooling cost advantage over cast-in-place processing.

With each of the principal elements of the windshield optimized and verified essentially in isolation, it was now time for the final verification of the concept; in other words, qualification of actual windshield hardware. The results of this extensive testing program are fully covered in Sierracin/Sylmar Test Report ER-72-006 (see list of references). Figure 15 shows the type of testing conducted for this program, and a brief summary will be provided in this text.

1. As a part of the acceptance criteria, each windshield assembly fabricated was subjected to two thermal shocks.

2. Humidity testing was conducted at 120° F and 95% relative humidity for 14 days and this same test was conducted at

140°F, and 160°F. All windshields passed.

3. Structural-deflection testing was conducted on a CH-46 windshield configuration on a contoured pressure vessel, and tested beyond the specification requirements through faceply failure, which occurred at 140% of ultimate wind-loading. With the glass faceply broken, this windshield proved itself fail-safe by withstanding 120% of the ultimate wind-load without failure of the plastic inner ply. The deflection monitoring set-up of the dial gages is shown in Figure 16, and the results shown in Table 3.

4. The endurance testing requirement is 2,000 cycles, from 35°F surface temperature to operating temperature, which is 110°F. We cycled the optimum CH-46 configuration a total of 6,000 cycles, or three times the required number, with no signs of deterioration.

5. Two CH-47 optimized windshields were fabricated for AVSCOM at no charge for flight tests at Fort Rucker, Alabama. The first unit had flown 990 hours, more than three times the qualification requirement, when it was inadvertently damaged by a severe impact during the installation of a co-pilot seat. The second unit is still flying with no signs of deterioration. (Reference Figure 17)

The end result of all this development and testing is the Sierracin optimum helicopter windshield, shown in cross section in Figure 18, as currently manufactured for Boeing-Vertol CH-46, and Boeing-Vertol UTTAS YUH-61A, shown in Figure 19, and with edge attachment variations such as those shown in Figure 14 is available to any other helicopter. This safest, lightest, most durable (Reference Figure 20), anti-icing windshield is currently service proven and available. Unfortunately for you who fly helicopters, the existing specifications do not demand the safety features of these new windshields. Inasmuch as these improved parts cost slightly more than all-plastic or all-glass parts, and all spares for existing helicopters are bought from the lowest bidder, there is little hope of their being widely used on helicopters now flying. Hopefully,

the benefits of this general fail-safe construction will be specified into future generations of helicopters, but should, in our view, be retroactively written into existing aircraft procurement documents.

The ultimate optimum helicopter windshield, that is, a glass faced polycarbonate fail-safe membrane unit with an inherently compatible adhesive interlayer, serves as a good spring-board to a new concept which would use a thicker polycarbonate inner ply and provide some degree of bird-proofing. This is one design generation further removed, but again, advancing material technology makes it a practical consideration now. Figure 21 is a cross section of just this type windshield without the desired degree of bird-proofing, but with superior spall and fail-safe characteristics, as previously mentioned.

List of References

Sierracin Glass/Plastic Composite Windshield
by: George L. Wiser, Sierracin Corporation

Status of Program to Develop Composite Windshields Utilizing Plastics and Chemically Strengthened Glass
by: Keith Gunnar, Sierracin Corporation

Anti-Icing Aspects of Helicopter Windshield Design
by: Phillip A. Miller, Sierracin Corporation

Helicopter Windshield Design Improvement Program
Engineering Report -- ER-72-006
by: T. R. Stefancin, Sierracin Corporation

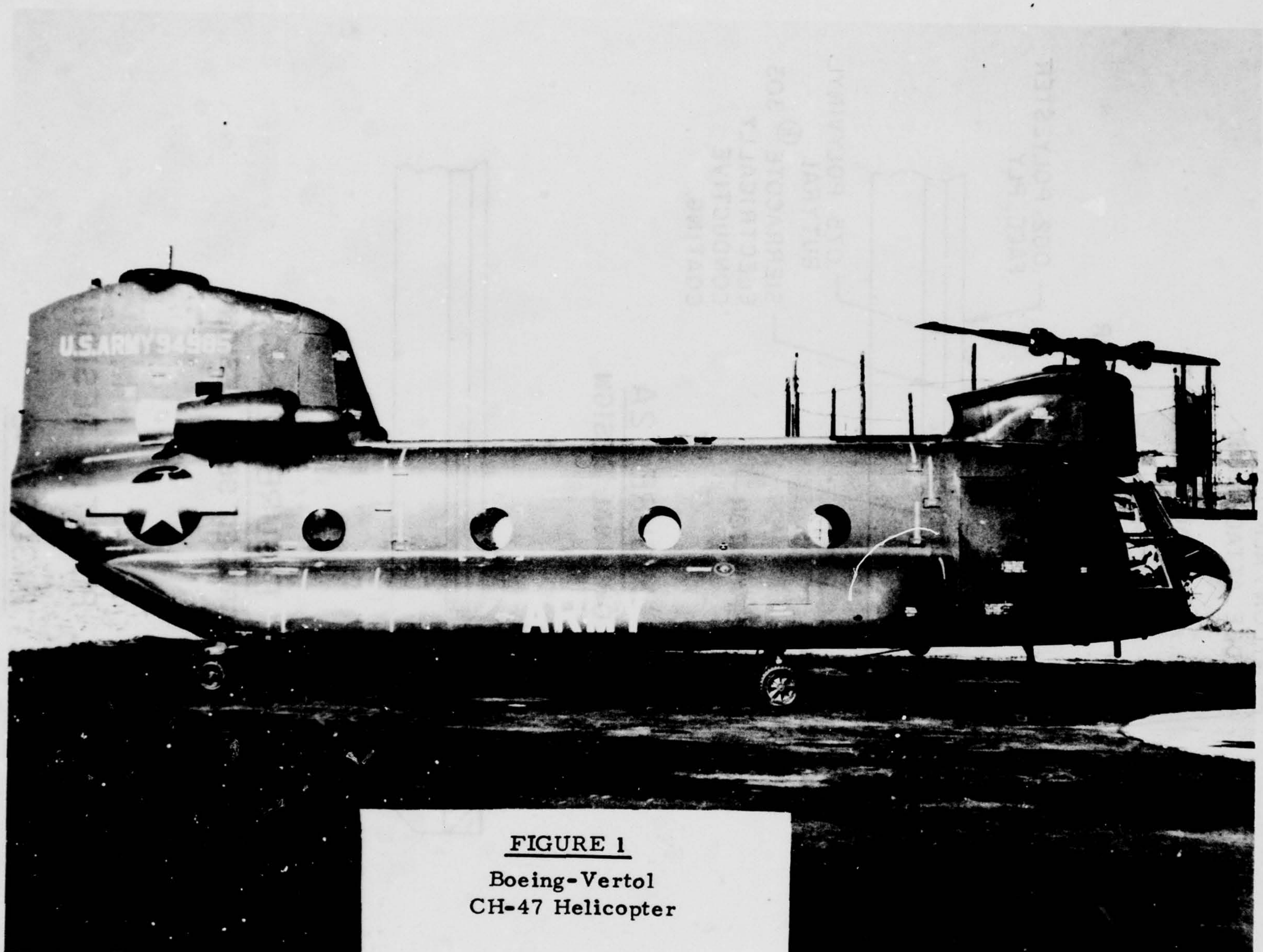


FIGURE 1

Boeing-Vertol
CH-47 Helicopter

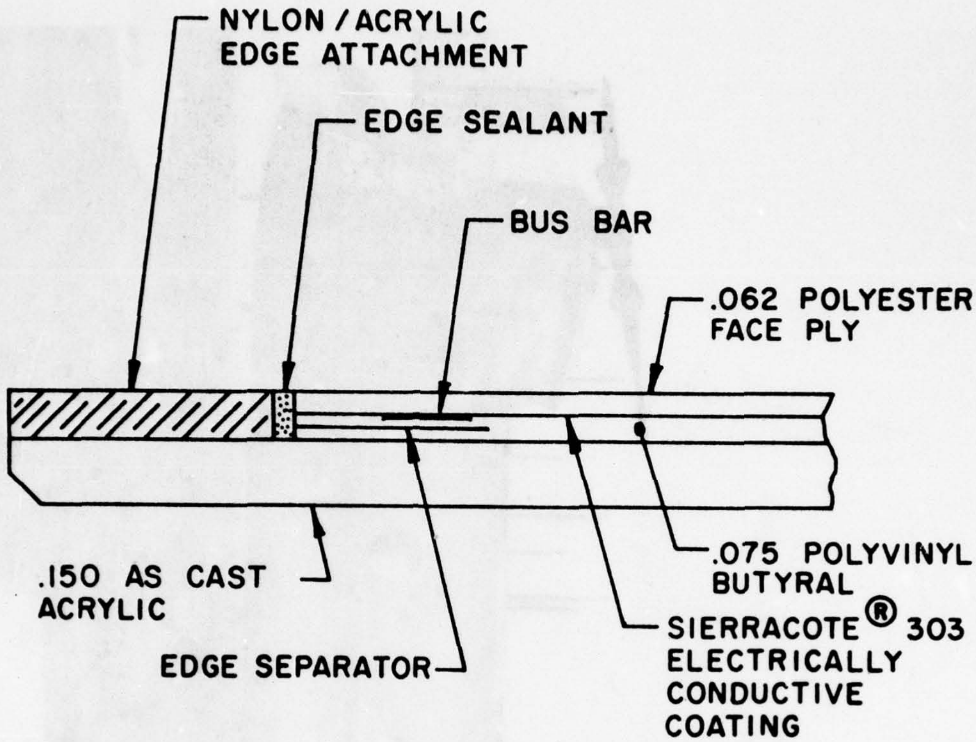


FIGURE - 2A
ORIGINAL DESIGN

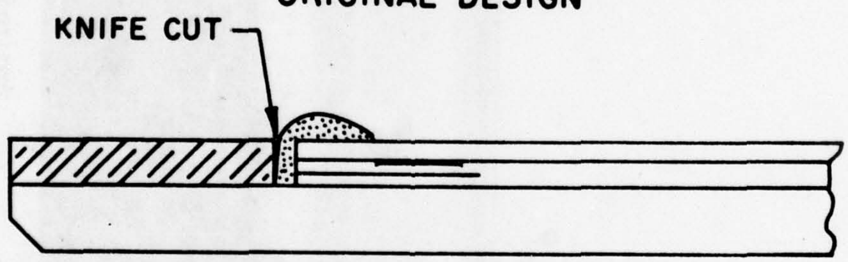


FIGURE - 2B
UP-GRADED SEAL DESIGN

ALL PLASTIC DESIGN

FIGURE - 2

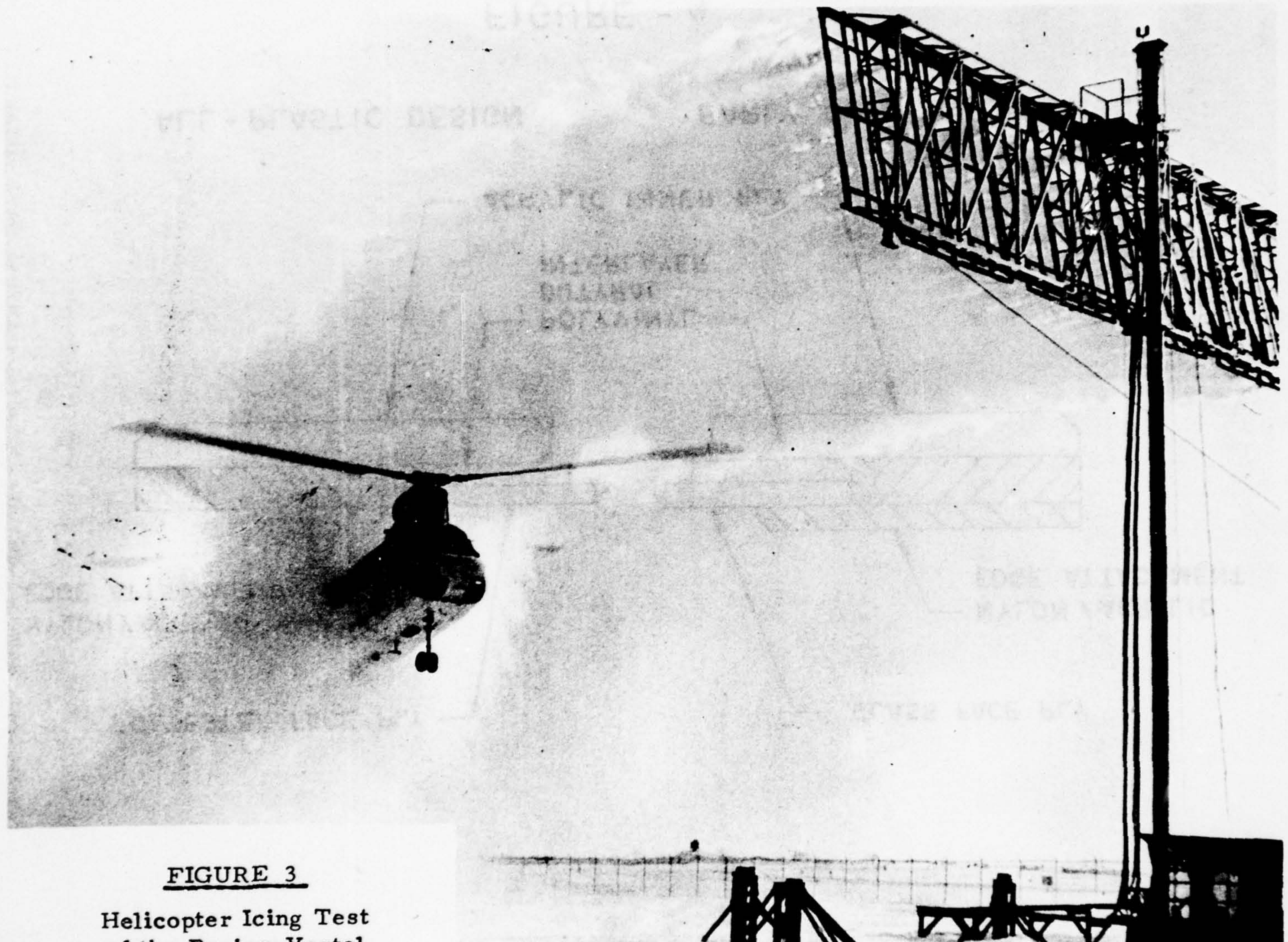


FIGURE 3

Helicopter Icing Test of the Boeing-Vertol CH-46 Helicopter

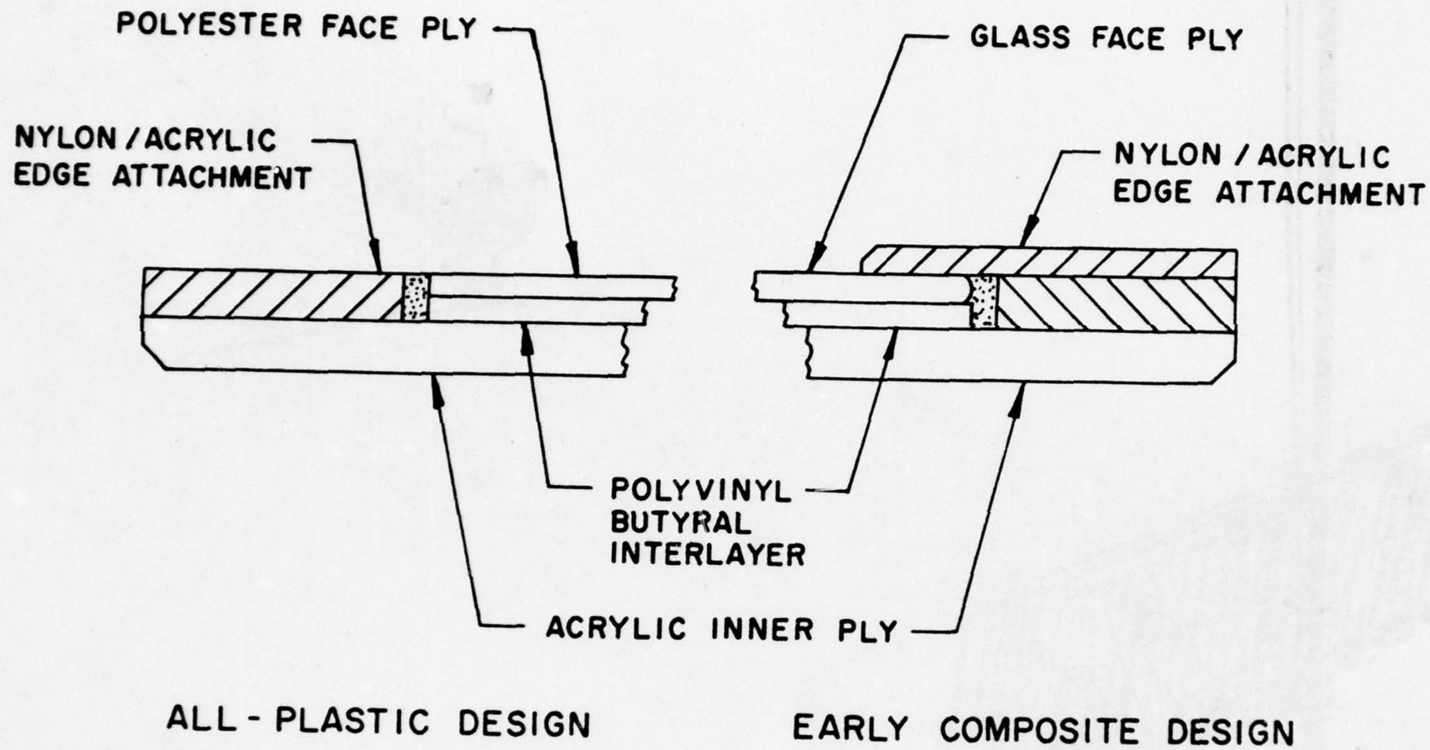

POLYESTER FACE PLY
NYLON / ACRYLIC
EDGE ATTACHMENT
GLASS FACE PLY
NYLON / ACRYLIC
EDGE ATTACHMENT
POLYVINYL
BUTYRAL
INTERLAYER
ACRYLIC INNER PLY
ALL - PLASTIC DESIGN
EARLY COMPOSITE DESIGN

FIGURE - 4

TABLE 1

BALLISTIC/SPALL/VISIBILITY TEST SPECIMENS

Specimen Cross Sections:

Specimen No. 1 (All-Plastic)

Faceply	--	.06 thick Sierracin S-900
Interlayer	--	.075 thick PVB
Inner Ply	--	.150 thick As-Cast Acrylic

Specimen No. 2 (Composite)

Faceply	--	.085 thick 0401 Chemcor
Interlayer	--	.075 thick PVB
Inner Ply	--	.125 thick Stretched Acrylic

Specimen No. 3 (Composite)

Faceply	--	.085 thick 0401 Chemcor
Interlayer	--	.075 thick PVB
Inner Ply	--	.150 thick As-Cast Acrylic

Specimen No. 4 (All-Glass)

Faceply	--	.085 thick 0401 Chemcor
Interlayer	--	.075 thick PVB
Inner Ply	--	.085 thick 0401 Chemcor

Specimen No. 5 (Composite)

Faceply	--	.085 thick 0401 Chemcor
Interlayer	--	.075 thick PVB
Inner Ply	--	.060 thick Polycarbonate

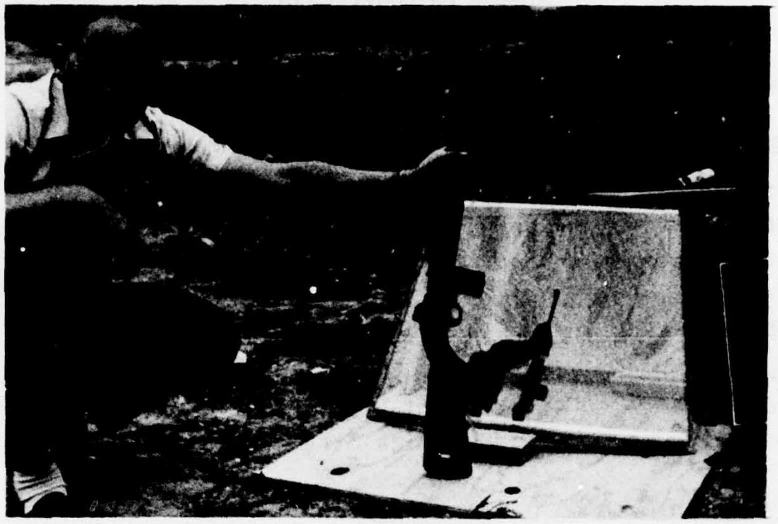
Specimen No. 6 (Composite)

Faceply	--	.085 thick 0401 Chemcor
Interlayer	--	.075 thick PVB
Inner Ply	--	.080 thick Stretched Acrylic

Spall Witness

1.0 thick styrofoam faced with aluminum foil -- 3/4 inch thick plywood sheet was used as back-up for styrofoam.

WINDSHIELD PRIOR TO
BALLISTIC PENETRATION



WINDSHIELD AFTER PENETRATION
WITH TWO 30 CALIBER BALL PROJECTILES

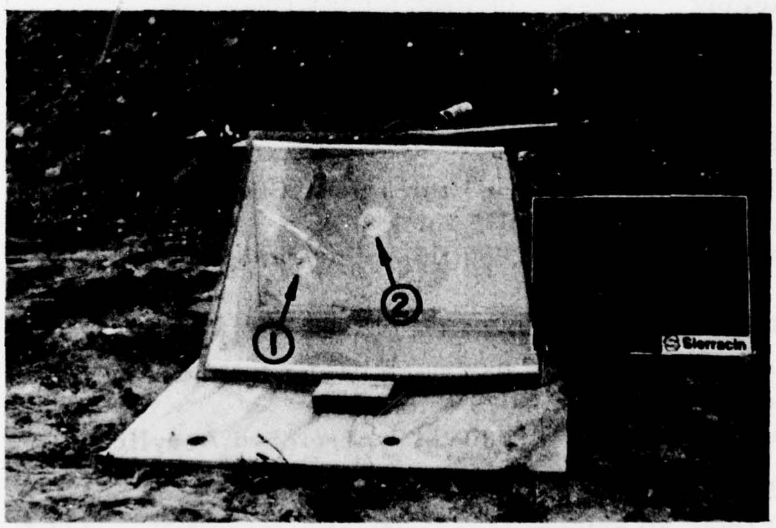


FIGURE 5

FIGURE 6

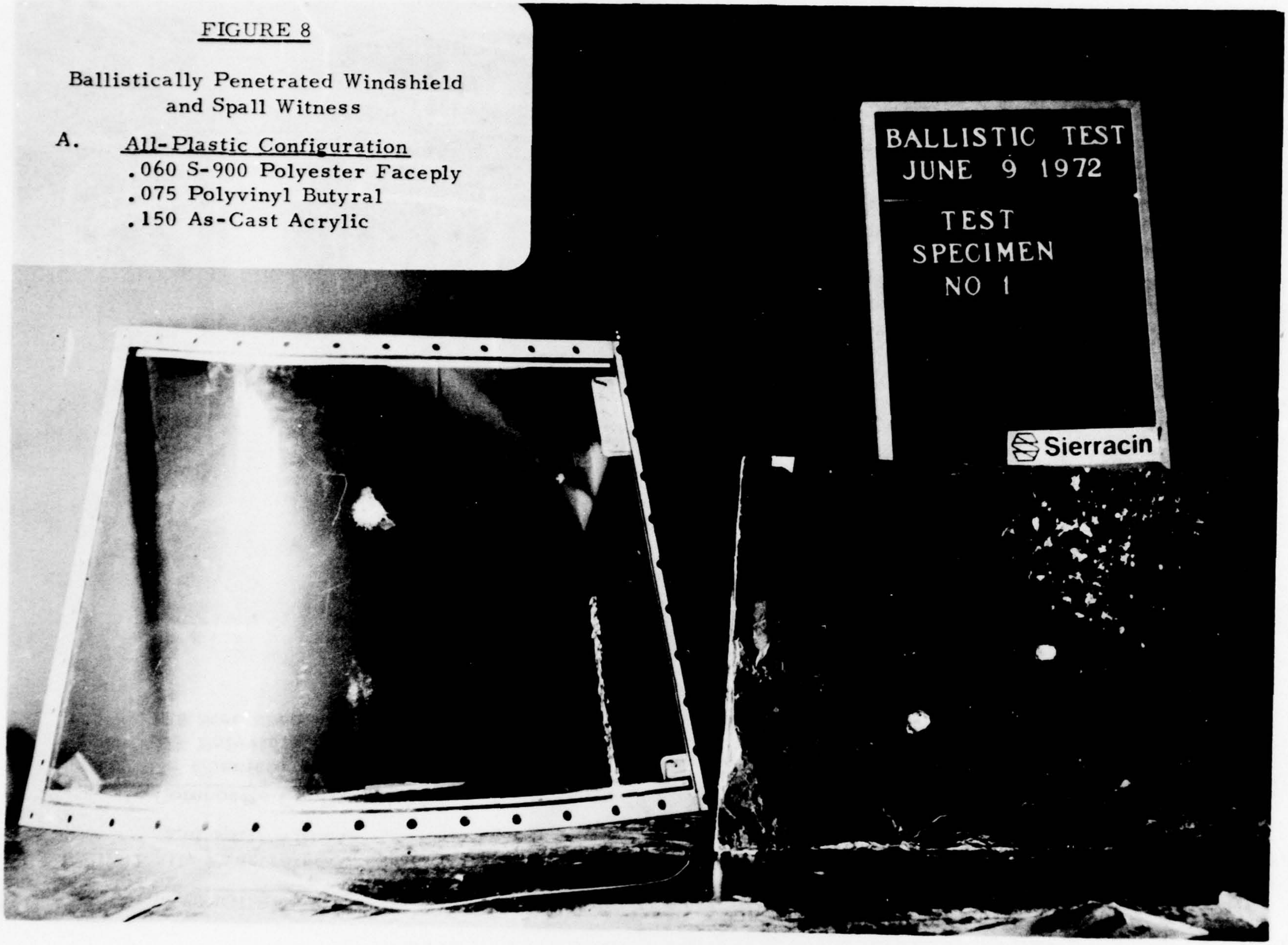
Test Set-up for
Fail-Safe Wind Load Tests

FIGURE 7
Fail-Safe Wind-Load Test
After Ballistic Penetration
Showing Failure of the
All-Glass Windshield

FIGURE 8

Ballistically Penetrated Windshield and Spall Witness

A. All-Plastic Configuration
.060 S-900 Polyester Faceply
.075 Polyvinyl Butyral
.150 As-Cast Acrylic



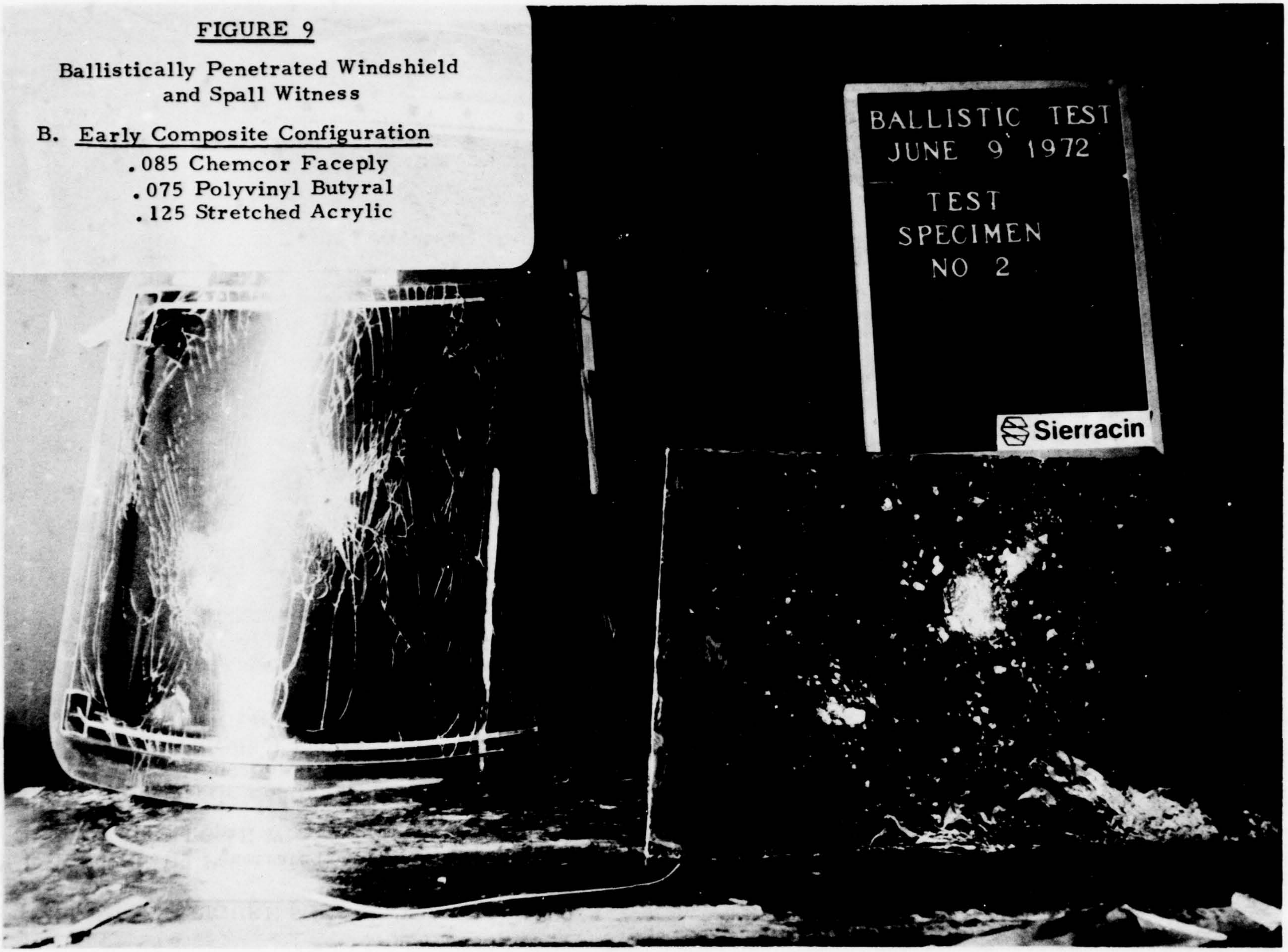


FIGURE 9

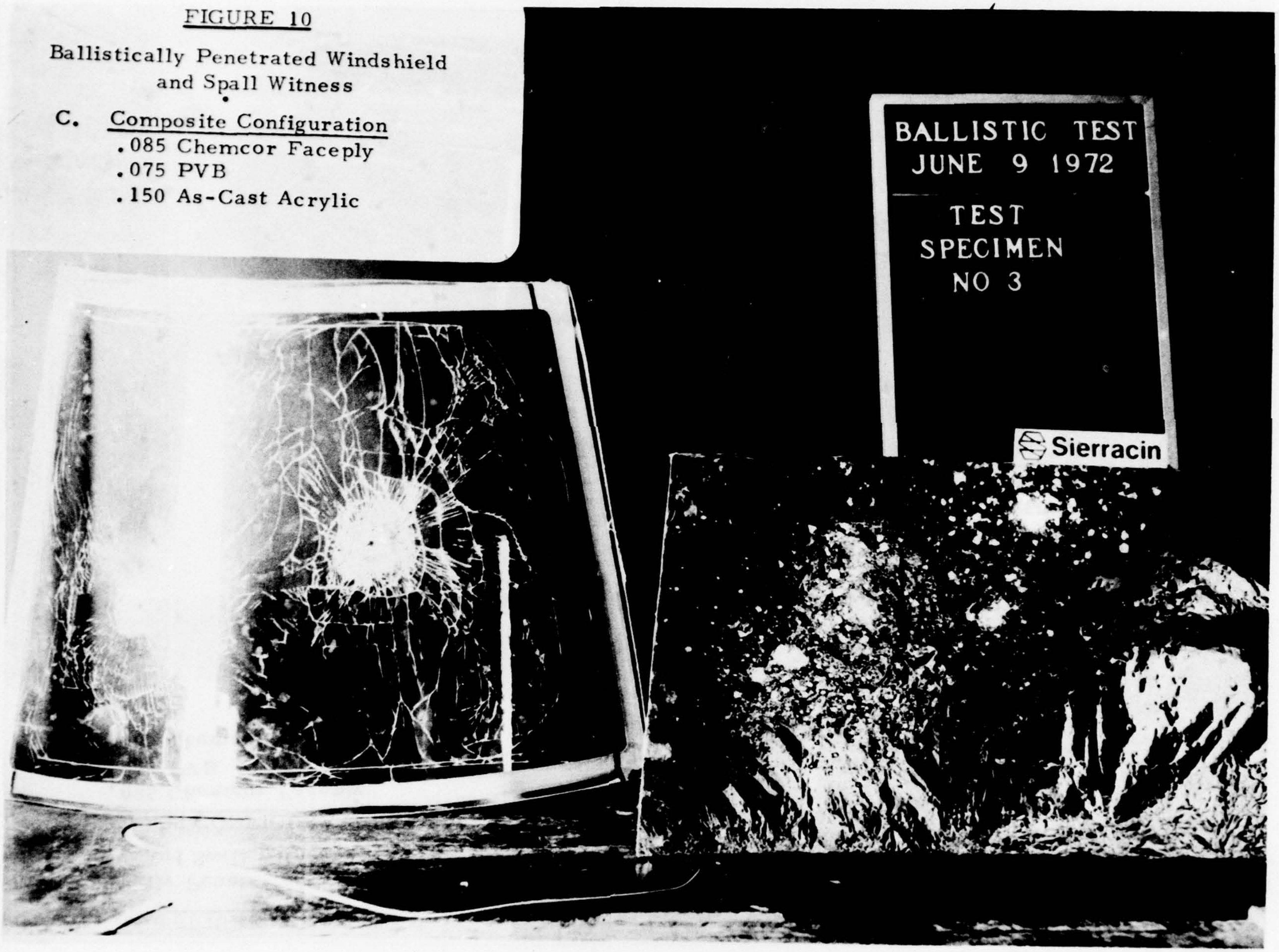
Ballistically Penetrated Windshield and Spall Witness

B. Early Composite Configuration
- .085 Chemcor Faceply
- .075 Polyvinyl Butyral
- .125 Stretched Acrylic

FIGURE 10

Ballistically Penetrated Windshield and Spall Witness

C. Composite Configuration
- .085 Chemcor Faceply
- .075 PVB
- .150 As-Cast Acrylic



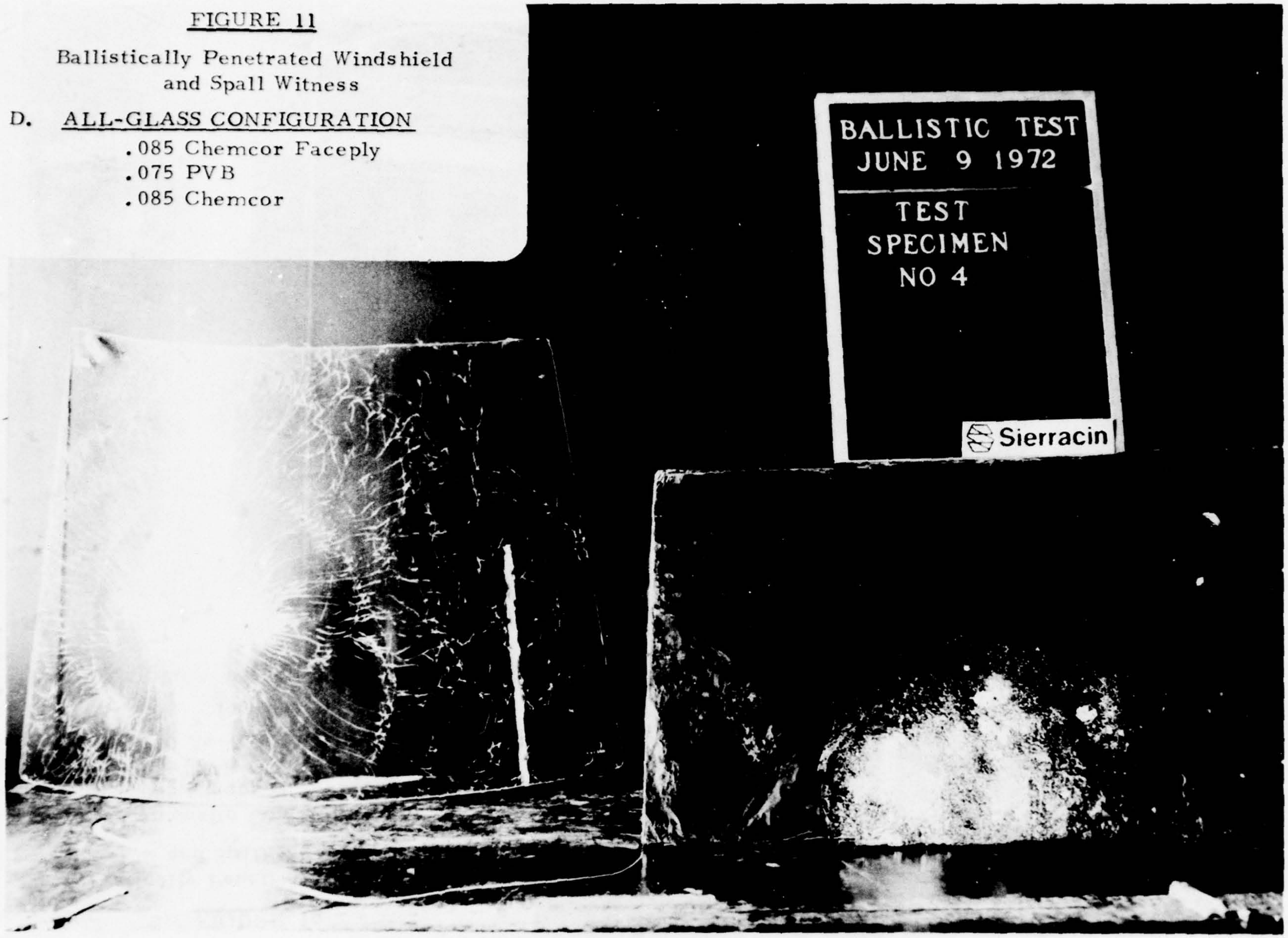


FIGURE 11

Ballistically Penetrated Windshield and Spall Witness

D. ALL-GLASS CONFIGURATION

.085 Chemcor Faceply
.075 PVB
.085 Chemcor

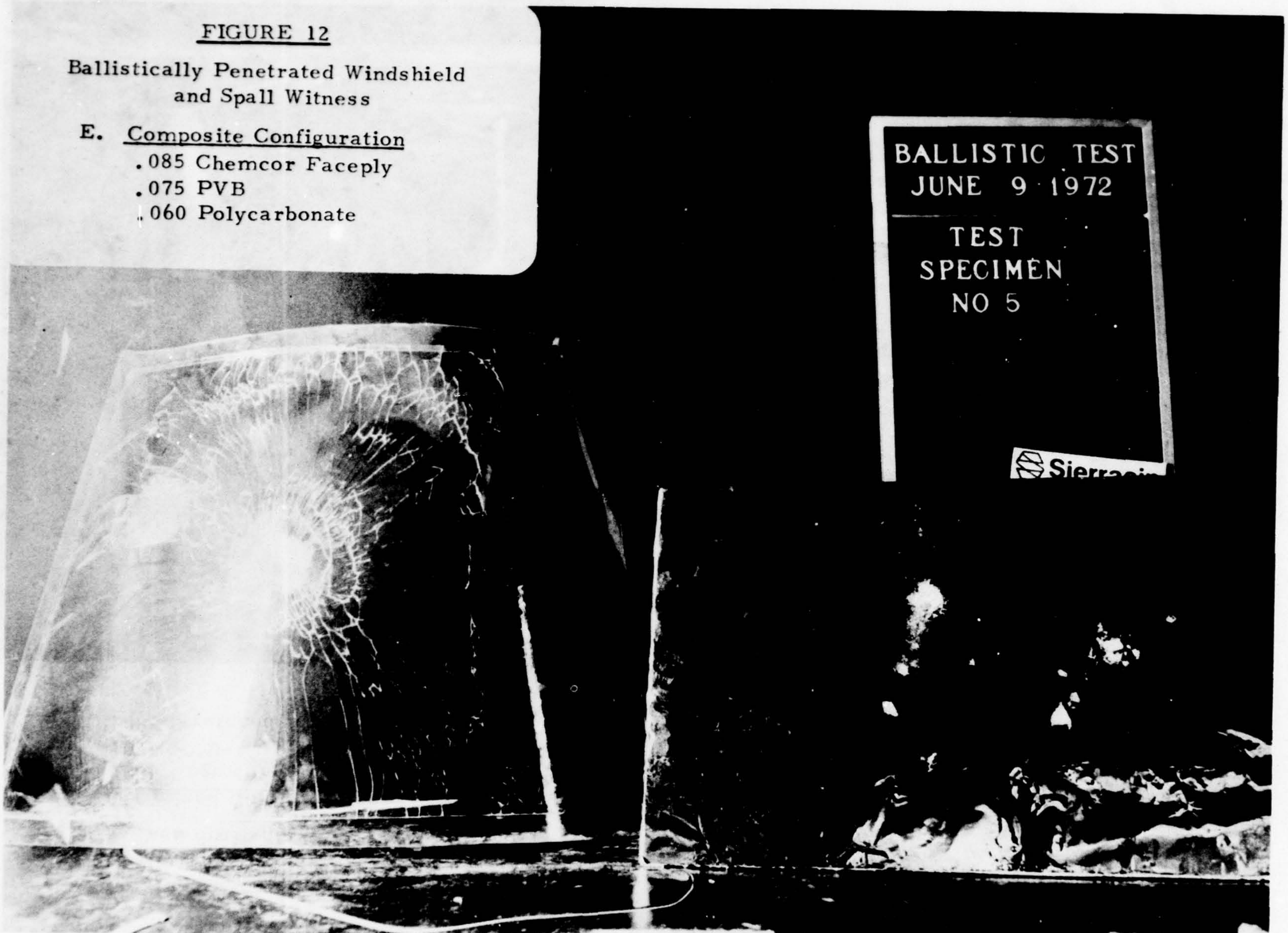


FIGURE 12

Ballistically Penetrated Windshield and Spall Witness

E. Composite Configuration
.085 Chemcor Faceply
.075 PVB
.060 Polycarbonate

FIGURE 13

Ballistically Penetrated Windshield
and Spallshield

F. Optimized Configuration
.085 Chemcor Faceply
.075 PVB
.085 Stretched Acrylic

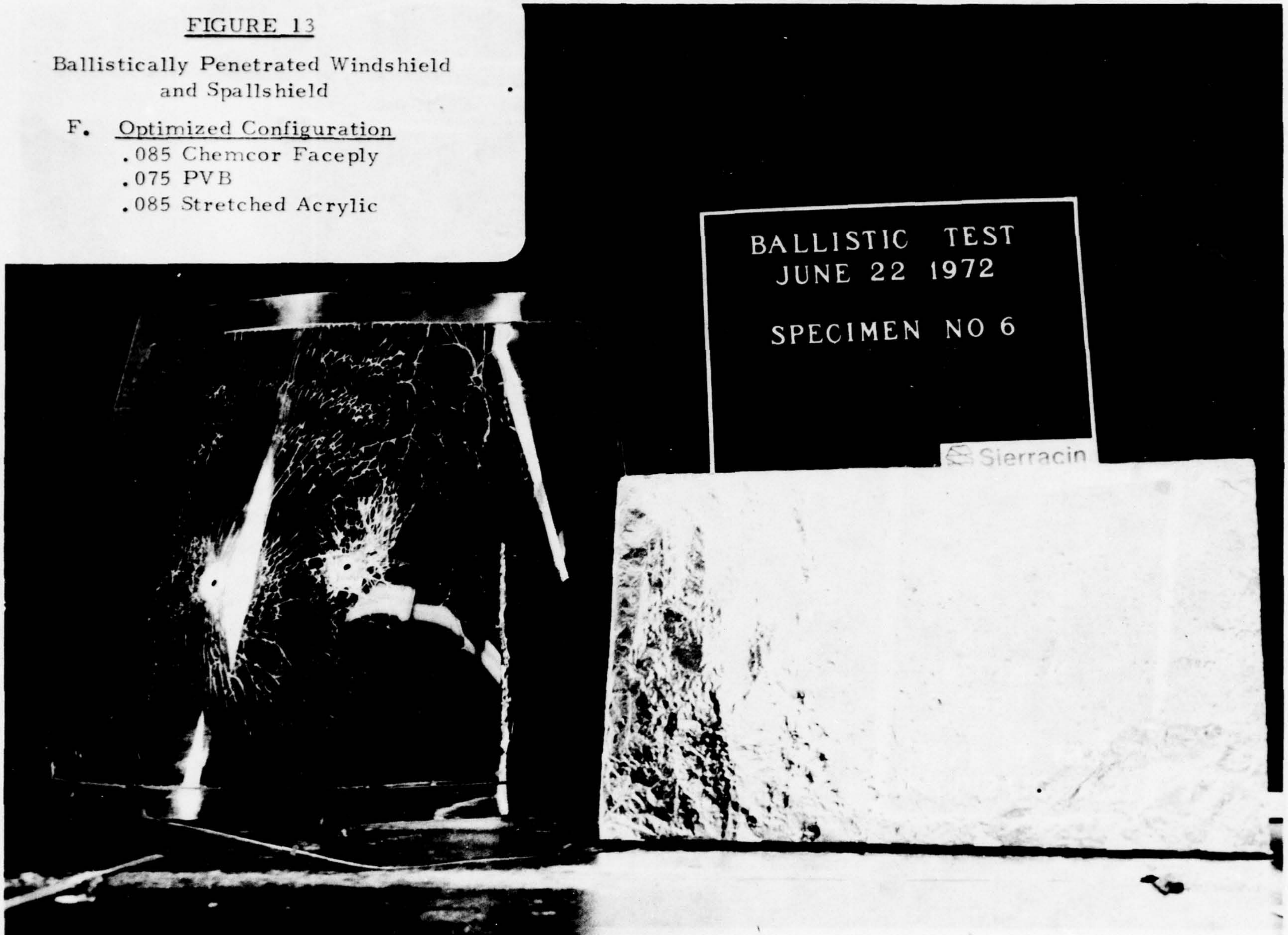


TABLE 2

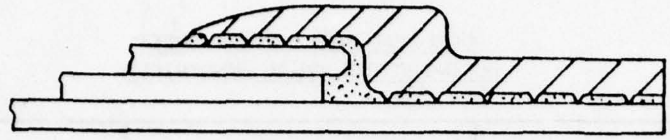
CONFIGURATION SUMMARY SHEET
BOEING-VERTOL CH-46 WINDSHIELD

	CROSS SECTION	ESTIMATED WEIGHT & CHANGE RELATIVE TO BASELINE 1	BALLISTIC SPALL TEST 2	FAIL-SAFE WIND-LOAD TEST 2
A	ALL-PLASTIC CONFIGURATION .060 S-900 POLYESTER .075 POLYVINYL BUTYRAL .150 AS-CAST ACRYLIC	REF. WEIGHT 12.75 LBS.	PASSED	PASSED
B	EARLY COMPOSITE CONFIGURATION .085 CHEMCOR FACEPLY .075 POLYVINYL BUTYRAL .125 STRETCHED ACRYLIC	BASELINE WEIGHT 16.85 LBS. SHIPSET WEIGHT 33.16 LBS.	PASSED	PASSED
C	COMPOSITE CONFIGURATION .085 CHEMCOR FACEPLY .075 PVB .150 AS-CAST ACRYLIC	17.33 LBS/ASSY. + 4.5% SHIPSET CHG. + 1.50 LBS.	PASSED	PASSED
D	ALL-GLASS CONFIGURATION .085 CHEMCOR FACEPLY .075 PVB .085 CHEMCOR	19.96 LBS/ASSY. + 20.4% SHIPSET CHG. + 6.76 LBS.	PASSED	FAILED CATASTROPHIC-ALLY
E	COMPOSITE CONFIGURATION .085 CHEMCOR FACEPLY .075 PVB .060 POLYCARBONATE	13.28 LBS/ASSY. - 20% SHIPSET CHG. - 6.60 LBS.	PASSED - SUPERIOR PERFORMANCE	PASSED
F	OPTIMIZED CONFIGURATION .085 CHEMCOR FACEPLY .075 PVB .080 STRETCHED ACRYLIC	14.65 LBS/ASSY. - 11.6% SHIPSET CHG. - 3.86 LBS.	PASSED	PASSED

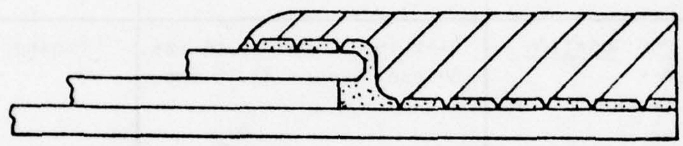
1 .125 THERMALLY TEMPERED GLASS APPROXIMATELY 3.35 LBS/ASSY. HEAVIER.

2 SEE SIERRACIN REPORT ER-72-006 FOR DETAILED TEST REPORT.

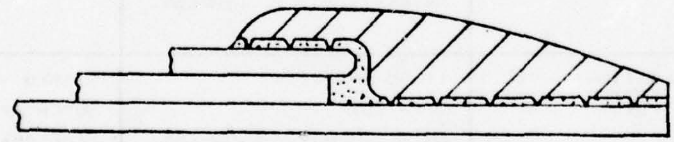
3 CONFIGURATIONS C THROUGH F ASSUME EXTRUDED THERMOFORMING EDGE REINFORCEMENT.



BOEING / VERTOL CH-46



BOEING / VERTOL UTTAS



PROPOSED SURFACE MOUNTING

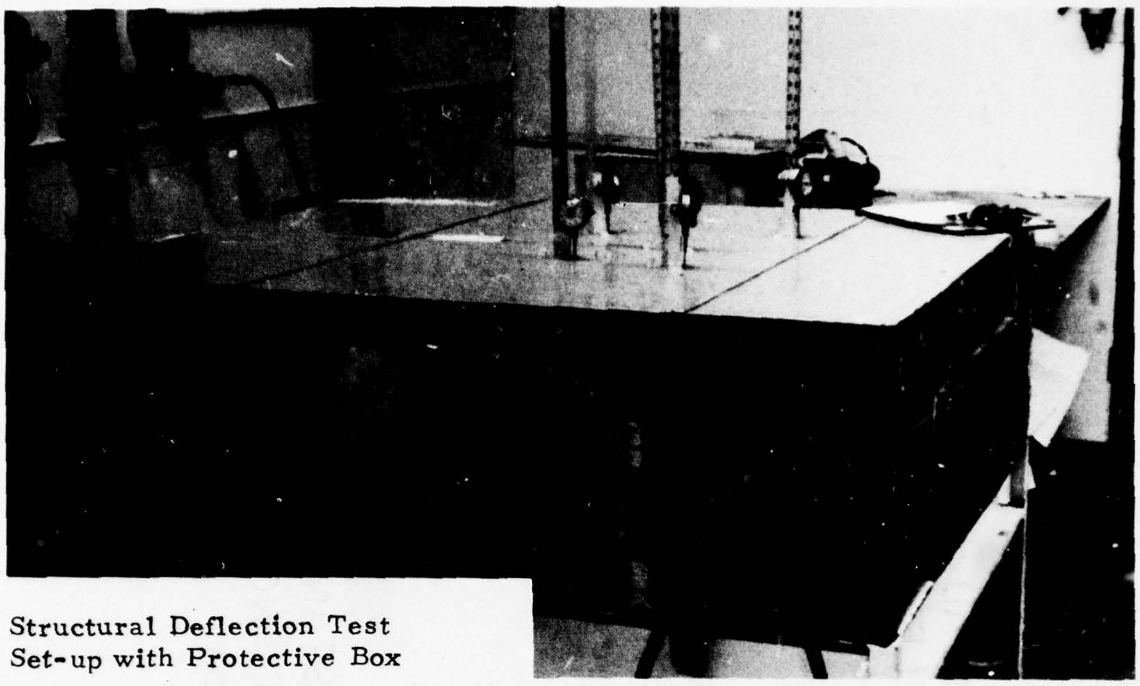
OPTIMIZED COMPOSITE CONFIGURATION

FIGURE-14

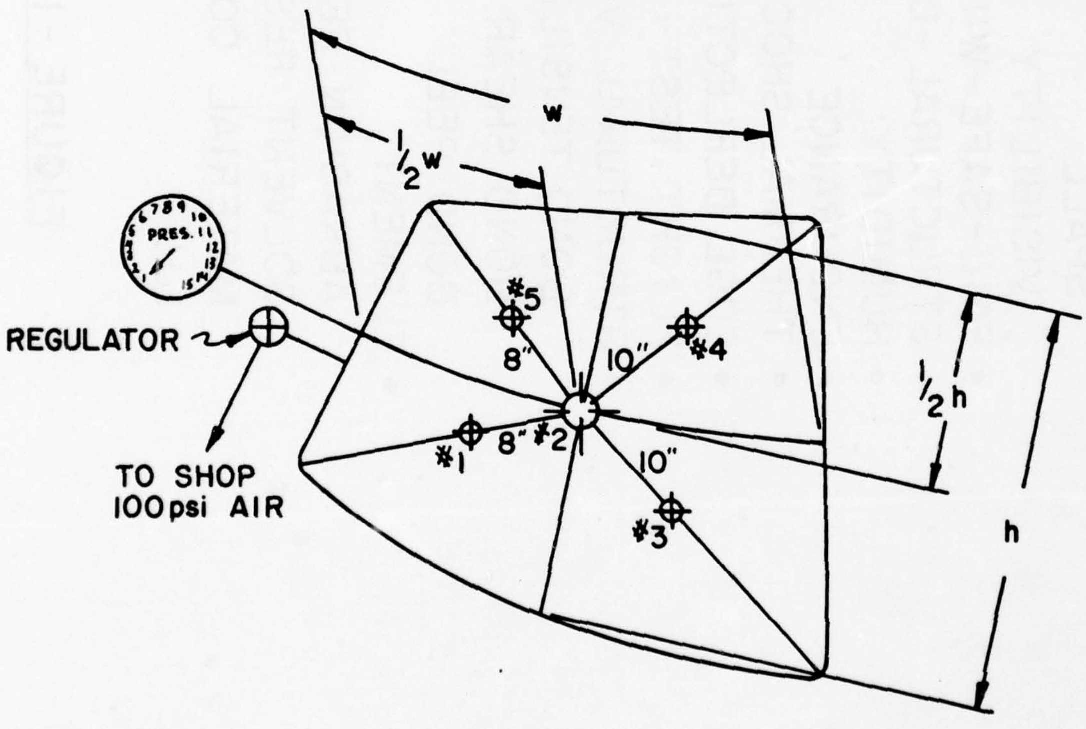
TESTING CONDUCTED

- BALLISTIC
  - SPALL
  - VISIBILITY
- FAIL-SAFE-WINDLOAD
- STRUCTURAL-DEFLECTION
- HUMIDITY
- ENDURANCE
- THERMAL SHOCK
- EDGE DEFLECTION
- FLIGHT TEST-FORT RUCKER CH-47
- STRUCTURAL VERIFICATION
  - BOND TENSILE
  - BOND SHEAR
  - BOND PEEL
- ELEMENT
  - ABRASION RESISTANCE
  - SOLVENT RESISTANCE
  - MATERIAL COMPATIBILITY
  - UV

FIGURE - 15



Structural Deflection Test
Set-up with Protective Box



PLAN VIEW SHOWING
GAGE PLACEMENT

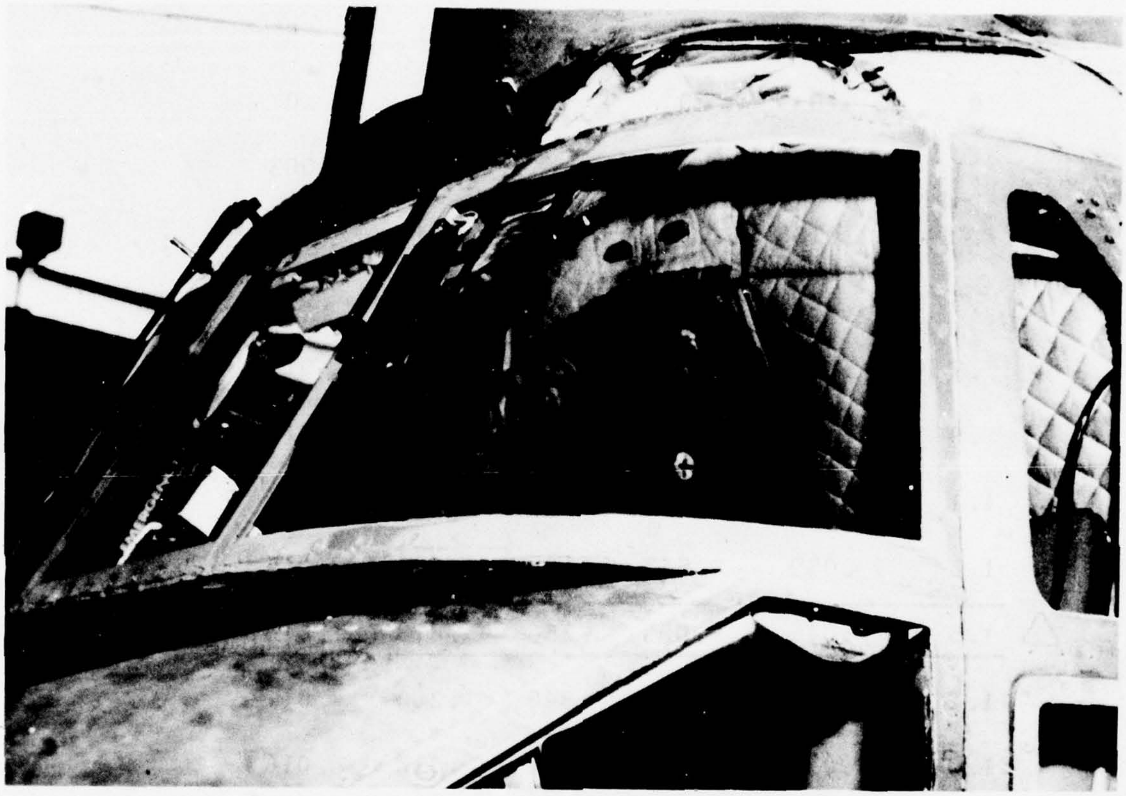
FIGURE 16

TABLE 3

CH-46 Optimized Design Configuration

	PSI	1	2	3	4	5
	0	0	0	0	0	0
	.1	.007	-.004	.025	.018	.003
	.3	.0135	-.009	.0485	.044	.0065
	.5	.022	-.018	.086	.088	.010
	.7	.027	-.024	.113	.113	.012
	.9	.030	-.029	.134	.134	.013
	1.1	.025	-.035	.162	.158	.014
	1.3	.039	-.043	.195	.184	.015
△	1.5	.045	-.055	.256	.218	.016
	1.7	.048	.060	.340	.266	.017
	1.9	.059	-.061	.401	.312	.016
	2.1	Pressure reached but gage reading could not be taken prior to faceply failure and loss of pressure. Pressure dropped to 1.8 psi and stabilized. This pressure was maintained for a period of one hour as a fail-safe test.				

△ Specification requirement for ultimate loading.



Optimized Windshield Installed On A
Boeing-Vertol CH-47 Helicopter for
Flight Testing At Fort Rucker, Alabama

FIGURE 17

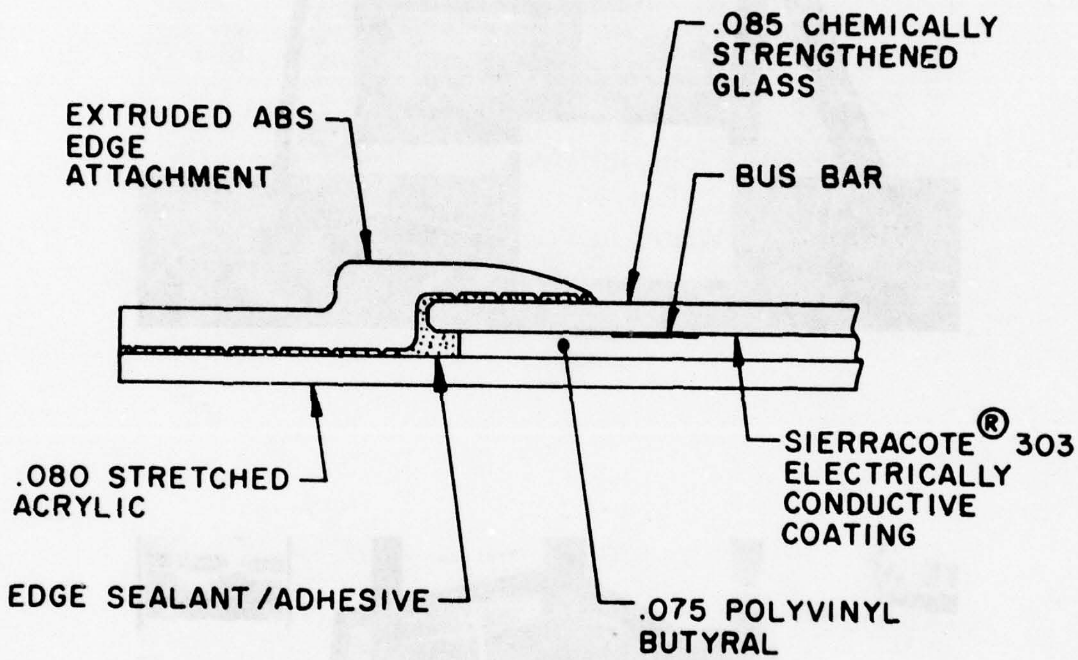


FIGURE - 18

OPTIMIZED COMPOSITE DESIGN


OPTIMIZED
CH-46
WINDSHIELD

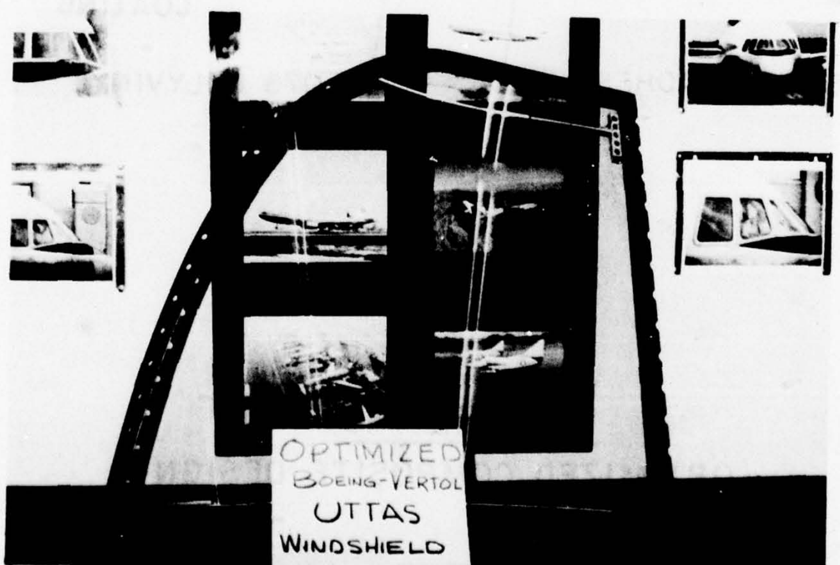
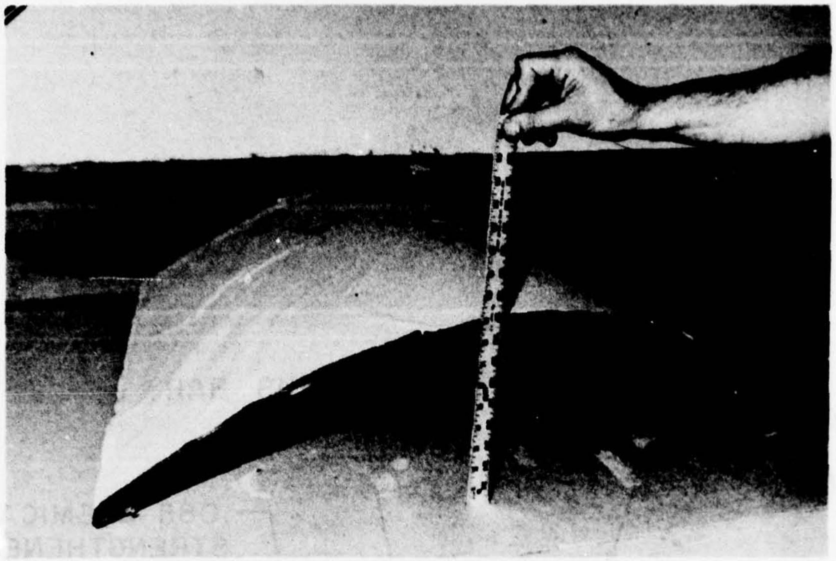

OPTIMIZED
BOEING-VERTOL
UTTAS
WINDSHIELD

FIGURE 19



Scale Indicates Free-State Stand-Off
of a CH-46 Optimized Composite Windshield
As 8.5 Inches

The 150 pound man shown standing on the windshield, in the free-state, deflected this windshield 3.0 inches without damage.

FIGURE 20

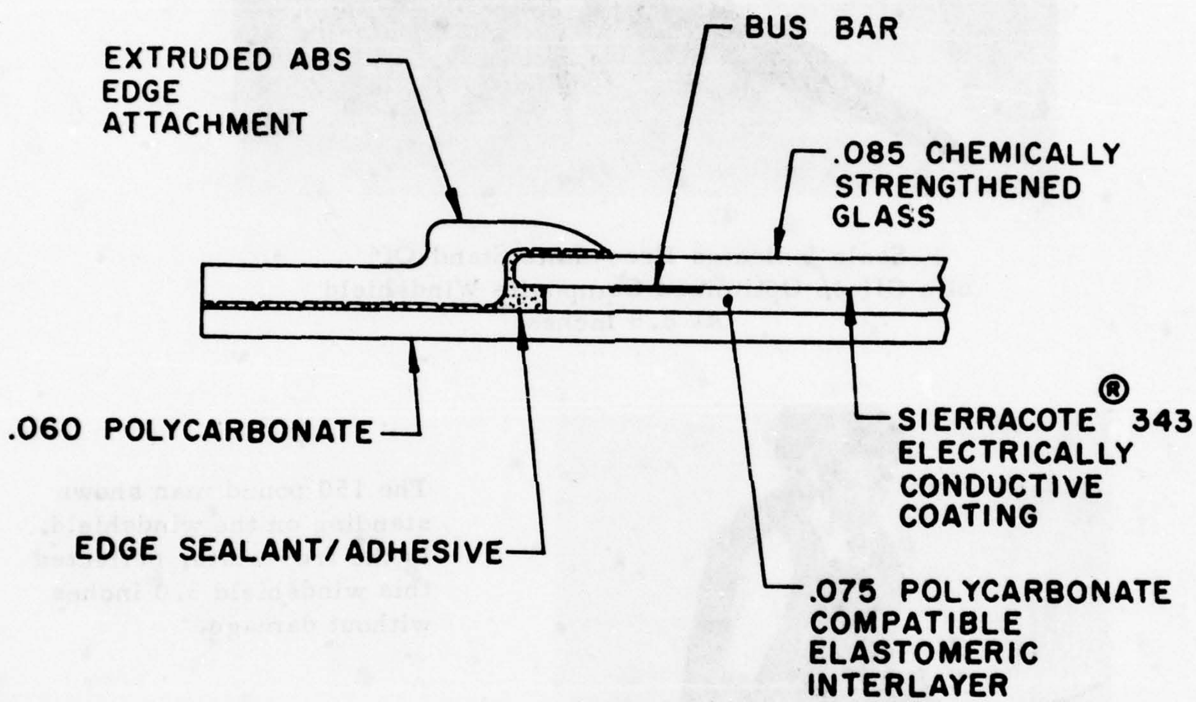


FIGURE-21

UP-DATED OPTIMIZED COMPOSITE DESIGN

DISCUSSION

LTC Graham, 2.75 System: I suggest that the lack of a reliable forecasting capability is more grave than you state; because it is the basis on which a tactical commander either commits or does not commit his troops; and he's doomed in both directions if the weather is predicted to be safe and it turns out to be unsafe, he's wrong; and if it's predicted to be unsafe and it's really safe, he's still wrong. So they should be formally tasked to improve the capability to our objectives. I think there are some alternatives even though we are formally committed. I believe, to get our weather service from the Air Force, there is an extensive capability that might be used in the form of ballistic meterology data that is available across the entire Army area for which there are literally hundreds of soundings made. The state of the art that's being advanced by the Atmospheric Science Laboratory to do these soundings by rocket and ballistic means, other than the conventional ballon, perhaps by adding the capability to detect and measure those quantities necessary to access icing would be just another delta in their effort.

Mr. Adams: I appreciate your comments; and I believe the ballistic soundings that you referred to are reported to the Air Weather Service; and this is included in their data bank and is included in the forecast. This is what I was told. I think that one of the ways to get around the problem for the operational commander is to ultimately have an onboard capability for knowing what kind of condition he is encountering. Now he can believe the forecast or not. If he has to perform the mission and he has a forecast moderate icing condition or a severe condition. If he has an onboard detection system, and if he has an ice protection system, he can deploy; and if then he finds out that the condition is too severe he can abort the mission. But at least he'll be able to deploy; and that's the thing we can't do right now.

Dr. Rosen, Sikorsky: I think you are quite correct. I think I liked your presentation on the liquid water content meter, or liquid water content temperature meter, or whatever you call it. I think you are quite correct in singling out those 2 parameters. We talked early that there are 3 basic parameters; liquid water content, droplet size and temperature. I think you are quite correct in singling out those 2 as the most important of the 3. The heart of the ice detection system or rate meter would be the ice detector, and those of us who have been through the ice detector business for 8-10 years and have seen all the excuses why it didn't work. O.K. Well you should have put it over here, don't you know, well, you have to turn it this way and, well, you put it over there you know you're going to get this kind of flow and, you know it gets a little tiresome after while. I must say that

the Lee Instrument concept, and I understand that Rosemount has a similar concept of creating a flow pattern over the probe and in my judgement solves the problem. In other words, what they're doing is creating an artificial situation where they are inducing air over the probe. Now I would hope that the development of the systems that you will be testing would have their own air flow capability so that we don't go chasing our tails on that. That you can make that available to us.

Mr. Adams: Most certainly that is the case with the Lee detector as you know.

Dr. Rosen: Well, is it the case with the Rosemount detector that you have?

Mr. Adams: There may be a Rosemount representative here, I don't know; but I do know they have developed within recent months, an aspirator to aspirate air over the detector, and whether or not this is at this point in time at a stage that it's going to be usable on our test aircraft or not we don't know; but we do intend to install the Rosemount detector in either configuration.

Dr. Rosen: I speak from personal knowledge, having held it in my hand, and I can tell you that it is available to you. You should definitely use it.

Mr. Adams: Yes, O.K. very good.

Unidentified, HLH Project Manager's Office: In listening to your briefing, I believe that your plan or approach for solutions was a long range type of program. Yet, I guess, my concern is that I see the next generation of future aircraft designs very rapidly becoming firm, for the Army that is, at least. I was wondering if you have given consideration in looking at this rapid design formation, and when having to convey some concrete results that will be able to influence these designs?

Mr. Adams: Well I'm not specifically familiar with the schedules for the production HLH program. I do know the production HLH does require ice protection of some sort. Of course at the Eustis Directorate, as you know, Bud Carper and his team are running the advanced technology component development program for the HLH and the prototype program and we're constantly in contact. It is a problem for any of our future generation aircraft. The UTTAS is coming right along, and the AAH, as well as the HLH, and we need to complete these efforts as quickly as possible and as the program goes along there will be results that can be revealed.

I think that these results will be useful to you, as HLH Program Manager, and to anybody else that needs the information. While we're in the process of finishing the program, we have to keep in mind that the results that we reveal are only partial results at that time; and we can give you as much confidence as we can in those results.

SQDN LDR Lake: The talk opens up the whole range of what we're trying to do and I'll just touch on 3 or 4. We can conclude having searched for many years for ice, that the present match forecasting system both in Europe and in North America is not even, doesn't even form the basis of discovering as to whether you're going to find it. The sensitivity of the environment to the development of ice, seems altogether out of the context of the ability of the matchman to predict temperature levels and humidity levels; and you can see its that if they can be wrong by a couple hundred miles they can be wrong by a couple thousand feet and it's all completely useless. (Garbled comments) This is, I think well known and should be pursued I think at great speed. What I am concerned about with this management concept of hitting the problem is that it looks great until you look at individual items. The bit that you haven't mentioned is that I haven't heard any manufacturer here say that he's going to give us any advice as to how much ounce to ounce force his rotor is going to be able to take in the way of mass, and how much degradation of his beautiful new leading edge type shapes that he is going to be rocketing around the country at 180 knots, he can stand before the rotor falls apart. If we're talking about the certification of aircraft we normally require wind tunnel tests, Category 1 and Category 2 trials which may last many thousands of hours, and as I see your program you are intending maybe to certify aircraft by the use of one airplane in a very difficult environment. How are you intending to certify the aircraft throughout its flight envelope while flying in natural icing?

This is an area that I think we really should put ourselves to. We have found it extremely difficult to find the conditions in, a safe manner, remembering that something bad might happen. We found it even more difficult to hold the condition, and that I mean plus or minus 200 feet a mile this way or that. Yet we seem to be proposing to certify an aircraft throughout its maneuver flight envelope with the ice on the aircraft, and if it's a rotor deiced aircraft we are by implication allowing a certain degradation of the airfoil before we clean it off, and this is an area that, I for one, am extremely concerned about. We've heard a lot about how to hack the problem from the organizational point of view; and we're setting up great programs but when you get down to

trying to do the things the way we normally do with an airplane, I haven't seen any suggestions yet.

Mr. Adams: Thank you for your comments. I may have misled you in regard to the qualification of future generation aircraft. It is intended, I believe, that the future aircraft such as UTTAS and AAH and HLH or whatever, will be certified for flight in icing conditions. Now specifically how this will be done we don't know yet. I think that what we anticipate is to perform this certification as we do now, basically under simulated icing conditions, under known environmental conditions with known liquid water content and temperature and droplet sizes, if we can measure those. This does not imply that we're intending, maybe I misunderstood you, that we intend to use this UH-1 test aircraft to certify the other aircraft. No, that's not the intent at all. Maybe I misunderstood you. Does this clarify your concern here at all?

SQDN LDR Lake: The concern again is that if we take this program, if you imply that this is a typical program that could be carried out on future aircraft. My concern is that the natural icing environment is difficult to hold, find and hold. It is extremely difficult, particularly with a fleet of one, to cover the operational flight envelope of the aircraft in any reasonable time. I think your program, and even your indicator, does assume that we will have a line to avoid which may be related to liquid water content and temperature. We may have a flight envelope which isn't so typical g velocity one; there might be great holes in it. It might be that susceptible to certain liquid water content, and certain temperatures, and yet we can go outside these and it will all still work. So the implication seems a little naive.

Mr. Adams: I'm not sure that I follow you, its a real problem here and I would like to discuss it with you later. I guess we are running short of time, Colonel Beasley, lets get together, Hugh, later and we'll talk about it.

Mr. Lazelle, Lucas: I was very pleased to hear Lockheed confirm some of the conclusions we came to when we designed the Wessex system, which is currently being evaluated by the British ministry as far as the control equipment is concerned. We also concluded it had many advantages to use a combined system and I would take issue when he stated that it was very difficult to grade heat along the blade for a spanwise heater. I think we've demonstrated on the Wessex and on the Bolkow and on many fixed wing aircraft that if you spray metal there is no real difficulty in grading the heat.

Mr. Schmidt: Well, in answer to that question we have found that the spray mat is not the best in the world as far as reliability is concerned and the same goes for warm wires. We considered the various alternatives for the heating elements which include photo chemical etching, piercing the elements, and then extending the sheets of metal. We also considered the spray mats and found from service experience, from the reliability record, that photo chemical etching offers the most reliable solution as far as heating uniformity, as far as longivity and as far as functional performance is concerned. I realize the spray mat may lend itself to variation of power intensity along the blades but we didn't use this method because of other considerations.

Dr. Rosen: I think the point made on spanwise watt density distribution and capability that the gentlemen from Lee Instruments made is quite correct. I think any of the available systems could be designed to give a variation watt density distribution. A couple questions - On photo etching have you considered the potential problems on fatigue and what results you'd have with that?

Mr. Schmidt: We are cycling the elements or samples of the chemical elements in the lab in a breadboard setup and sure, we are investigating the fatigue. I don't know how many cycles they have already accumulated, but so far we didn't encounter any problems. The photo chemical latching element has also been applied to the P3 and as I said during a whole year of perhaps 200 airplanes in service, we had only one spare set to change on the tail, so I think this speaks for itself. It is admitted that we don't have the fatigue problem on the fixed wing aircraft as on a rotor wing aircraft but that's why we are conducting our breadboard tests setup in the lab and we are cycling these samples and eventually we will have a good picture. However, based on the available data we know that our commercial aircraft applications didn't incorporate heater elements in the tail of some commercial aircraft about 10 years ago and these systems have been abandoned because the heater element didn't prove very reliable, there were

delaminations and other problems. So far we haven't encountered the problem.

Dr. Rosen: In your curve that showed power loss did you assume that the heater boot was flush with the surface of the air foil? In other words, were the power penalities that you displayed in your curve consistent with the aerodynamic penalities that would result on the blade and did they account for weight penalities in addition to the obvious electrical power penalities?

Mr. Schmidt: Yes, indeed, we did account for the weight penalities associated with applying the heat, not only for the weight penalty itself, but also for the fuel, additional fuel required to carry this weight penalty.

Dr. Rosen: Were you using a flush design?

Mr. Schmidt: Remember that the trade-off studies we conducted do not apply to the UH-1 helicopter that's going to be modified. We looked at the entire array of future helicopters from 4000 to 32,000 lbs. and it was a basic assumption that the heating system will be incorporated during the initial design of the system, so there was no special drag wise due to heaters that were not flush with the surface. That is not related to the current experimental program. The study was related to a broad range of vehicles and was based on the assumption that the heater element is incorporated right from the very beginning.

Dr. Rosen: Can you tell us how you obtained the percent chord coverage which you indicated was consistent throughout the entire span?

Mr. Schmidt: On the UH-1, we used 25 percent cord and 10 percent. We went thru the angles of attack that are characteristic for the UH-1; and we conducted trajectory analysis when the foil, air foil, was placed in the field of the impinging droplets. The impingement limits have been determined and these impingement limits are based on a 40 micron limit; whereas the design limit is usually 20 microns. The impingement limit is based on twice the normal design diameter of the droplets. This is on the maximum possible diameter in the clouds. If the cloud, let's say is assumed to have a 20 micron average diameter, the impingement limits were based on a 40 micron diameter, and we have a computer program wherein we put in the coordinates of the shape of the air foil, then we are determining the limiting trajectories of the water droplets.

Dr. Rosen: Why are the trajectories, why do they yield the same results throughout the entire span?

Mr. Schmidt: They don't yield a constant chord along the span, we selected a compromise value appropriate for the entire span. To simplify design and keep the cost coverage constant we used the impingement limits that are associated with the furtherest aft point of impingement.

Mr. Wilson: Just a supplementary question to that one. When you'd established the impingement limits and gone for this constant coverage spanwise, or rather constant cord wise coverage with span, did you take the inboard sections or the tip?

Mr. Schmidt: No, we used several stations, we used the profile of the blade at several span stations and we investigated not only at the middle station and the tip stations but at several stations. I think it was 4 or 5 stations and we made sure that the coverage is appropriate, and is compatible with the aft most impingement limit of the drop, so its not just for one station.

Unidentified: I was really intending to make a short comment if I could - It seems to me that possibly there's a body of statistical information that was not included in your summary resulted in the conclusion that the etched circuit resulted in greater reliability. This body of information would show that there is, running back to 1955, a great many hundred thousands of electrical deicing circuits which the manufacturer has never once been required to make a gratus replacement even though they were all guaranteed. They are wire elements; they are not elements based on printed circuits. Now that information is available from quite a variety of manufacturers who have used this particular type of installation very extensively. I could possibly after the meeting furnish some to you if you want it, but there's more available from many of the people here.

Mr. Schmidt: It is well taken, but our experience has been in this field, and now we are required to produce and we have been successful. Its hard to beat success. We don't want to try from A and then wind up with Z again. You have to remember it is an experimental program. We can't afford the huge amounts of money to test the warm wires versus the spray mat versus the pierced metal and stretch and then versus the edge metal. We relied on our experience and this experience has been good. We don't want to change horses in the middle of the stream; and we are not prejudicial, but we expended during the test, a lot of effort to make the system work; and it does work. And we know that with other systems you did encounter the delamination problems. They did abandon on some commercial aircraft, the tail deicing system. They had to certify by the FAA the tail without the icing provisions.

So we weren't going to go now into a grey area. I fully appreciate your comments and I'm sure they are valid and within a broader program that involved a larger scale and more effort and a greater degree of money you could compare side by side 4 or 5 different configurations of the heater elements. However, our motto is you can't beat success.

Ray Smith, AEFA: You made one statement there about the acceptability of the windshield used on our Huey. There was another part to this statement; it was acceptable within the scope tested. I might point out that the test was one flight at a $-6\ 1/2^{o}$ and further testing would be necessary to verify this for a range of temperatures and possible mixed conditions.

Mr. Dendy: I totally agree with you.

Unidentified, M.O.D. Navy, U.K.: In the program that we are currently embarked on in the U.K. for most of the British forces with heated screens, one of the criticisms raised by the test pilots, and also by the pilots in industry has been the disorientating effect either by installations in a curved windscreen or in a windscreen that is inclined at an angle to your line of view. Also I noticed in the picture yesterday in the UH-1 partially cleared screen, where you only had a strut heating in the screen, that your vision which you gained did not have either a true horizon in a horizontal sense or cutoff in a true vertical sense. In the already difficult conditions which one is flying in icing conditions, and the fact that your trying to look out, has any consideration been given to the disorientating effect one is going to get in these very poor visibility conditions?

Mr. Dendy: *The effort on the UH-1H thus far with the* aircon system employed across the face of the windshield was an initial effort, as stated just a moment ago. There are many other considerations. We are involved with Lockheed with totally heating this windscreen. As far as the buss bar location and disorientating effects and so forth, one of the things that bothers me; PPG is well aware of the many considerations in windshield design, when the air frame manufacturer comes to us they say, well we've got this shape and this size and its at this angle we have to accept it. One of the problems we're having is trying to get involved with the air frame manufacturer as soon as possible to get him to consider the shape, size, and angle of installation, so that we can work together. I believe there are a lot more considerations that the helicopter designer has to take into affect, initially, than just how to heat the windscreen. It's been assumed to date, (a perfect example of one of these things is the F-111 which was designed and flown with no consideration given to bird impact) they are retrofitting the fleet after for not taking this into consideration.

Mr. Schmidt: I would just like to comment on the visibility question - When we employed 3-phase power for our windshields the demarcation lines between the 3 phases do represent a definite

degradation of the visibility as far as the optics is concerned. Originally on the AH-56, we had 3 phase AC power and we went thru to the test program initially and the visibility problem became so acute that we had to go to DC power in order to eliminate the demarcation lines between the 3 phases. On this program that we are conducting now you are supplying one phase of power to the left hand panel, one phase to the right hand panel and the third phase of power to the buss on the main rotor; so that you have eliminated this demarcation problem.

Mr. Dendy: With regard to that, I could say that PPG is working on the phase separation lines. We've decreased them initially on some of the helicopters. I think they were up about 1/8 of an inch; they are now down to the area of .060 inch in width.

SIX YEARS OF FLIGHT TESTING AND DEVELOPMENT IN THE FIELD OF HELICOPTER ICING

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SUMMARY

This paper is a digest of the author's experience during six years of flight testing and development directed towards the major obstacle to achieving the 'All Weather Helicopter'. It considers atmospheric icing conditions and the ways in which ice can affect the helicopter. It deals with some of the problems encountered and the methods adopted to overcome these problems. It advocates the use of momentum separation, where this is a suitable method of protection in preference to the use of heat, which besides requiring a considerable amount of power, can, in certain cases, do more harm than good. Finally it concludes that, in the foreseeable future, there are likely to be occasions where immediate exit from icing conditions will be the only safe action to take. However improvements in meteorological forecasting, ice detection etc should eventually minimise the number of such occasions.

INTRODUCTION

During the last 15 years the helicopter has developed from a very simple aircraft with an extremely limited capability in fair weather to a sophisticated machine which can fulfil a number of operational roles involving IF flying. One sees advertisements in the technical press, glossy brochures etc extolling the merits of this or that 'All Weather Helicopter', but how many of these really can operate in all weathers? It is true that most of the present day helicopters can be flown safely in IMC conditions for prolonged periods, thanks to rotor governing, autostabilisation and various other refinements, but how long would they survive in icing, precipitating snow or what is even worse a mixture of both.

It is now six years since A&AEE was given the job of investigating the flying of helicopters in icing and snow. This was the first time that any serious attempt had been made to look at the overall problem in the UK. Prior to this, work in the helicopter icing field had been confined to the testing of engines and intakes in icing tunnels and trials of rotor de-icing systems in the NRC spray rig. Whilst some of these facilities are very good, the conditions which they produce are often very different from those produced by nature and therefore it is necessary to interpret the results in the light of experience of flying in the true environment; and here we come to the crux of the matter. What is the true environment? Well, gentlemen, having spent five winters and a considerable number of flying hours in search of icing I can assure you that it is usually much more complex than the icing environment defined in the CAA, FAA and similar specifications. The natural glaciation process which occurs during the life cycle of icing clouds gives rise to continually changing conditions. This can be further modified by snow precipitating from a higher cloud layer. Mr Abel dealt with this when he presented his paper on Wednesday morning but I would like to re-emphasise that the presence of ice particles in an icing cloud can have a very marked effect on the type of ice accretion on an aircraft and on the performance of thermal ice protection systems.

Once the pilot is committed to flying in cloud at temperatures below freezing, he has no means of foretelling the severity or complexity of the conditions he is likely to encounter. It is therefore incumbent upon those who are responsible to ensure that as far as possible when a release for flight in icing conditions is issued that the aircraft will survive the worst that nature can provide. In my opinion this can only be achieved by utilising the simulation facilities available to establish a level of confidence and then proceed to a protracted programme

of flight in natural icing conditions. This is the basis on which the A&AEE team have operated.

Before proceeding to the more detailed aspects I would like to state that our basic aim is to clear some of our turbine engined helicopters for continuous operation in layer type cloud up to 5000 feet at temperatures down to -20°C and in precipitating snow. Whilst deliberate penetration of cumulus cloud and flight in freezing rain is prohibited, it is accepted that these conditions will on occasions be inadvertently encountered.

As Captain Checketts has indicated, our approach to trials has allocated priorities to clearing the vulnerable areas of the helicopter as follows:-

first, engine protection

second, windscreens

third, ice detectors and other relevant instruments

and fourth, rotors.

and I would like to deal with these topics in that order.

ENGINE PROTECTION

The present trend towards top mounted engines with forward facing pitot intakes simplifies the engine/intake anti-icing problem. This type of intake is, however, very vulnerable to FOD in the form of ice shed from the main rotor or parts of the airframe forward of the engines. The first step therefore should be to shield the intakes from direct entry of FOD. This can be done in a number of ways such as the Sikorsky Ice Shield for the SH3D/ Sea King or the Mushroom Intakes as proposed by A&AEE. In the latter arrangement we argued that the face of the protecting surface should be convex so as to minimise ice accretion and impart radial acceleration to the oncoming water droplets, snow, ice particles etc, so that by their higher momentum they move across the airflow as it enters the intake radially behind the mushroom. Inevitably ice would accrete on the flares behind the mushroom and in the intake ducts if they were not heated because the momentum separation cannot be relied upon to give complete removal of the water and ice particles in the airstream. Testing on this form of engine protection and also on the Sikorsky Ice Shield have revealed that ice accretions which might result in engine damage can still occur and further development is

Slide 1 & 2

Slide 3

necessary.

Installations where the engines are buried in the fuselage usually necessitate some form of indirect entry involving at least two changes of direction of the airflow between the point of entry in the fuselage and the engine compressor. If a ducted intake is used, heavy ice deposits may occur at the bends due to radial acceleration of the water droplets causing local high impingement areas. If heat is applied only to these areas to prevent the formation of ice, the problem is merely transferred to the unheated portion further along the duct. In order to ensure complete freedom from ice under all operating conditions it becomes necessary to apply heat throughout the entire length of the duct. Apart from the fact that the amount of heat required is probably prohibitive, this can produce problems when flying through ice crystals which would normally pass through the engine without causing damage. If the heat input to the ice crystals is less than the latent heat necessary to effect complete liquification, ice will form on the heated surfaces and continue to grow until, by virtue of providing its own thermal barrier, sufficient heat is available on the surface of the duct to destroy the adhesion. Slide 4.

Let us now consider how we can avoid these problems. If we remove the constraint on the airflow by providing an expansion chamber between the point of entry on the fuselage and the engine, this will remove the localised high impingement areas. Therefore, the first step is to provide a plenum chamber with a good surface finish and free from internal obstructions. It is then necessary to ensure that the entry conditions at the plenum intake are the best possible. If the intake faces forward it will suffer from direct entry of snow, ice crystals and water droplets in forward flight. If it faces upward it will be subject to similar particles entrained in the rotor downwash during hovering flight or transition to the hover. Twin rearward facing intakes symmetrically disposed on each side of the fuselage is one possible solution, but they could suffer from aerodynamic distortion from the rotor downwash. This leaves us with the downward facing intake which appears to offer the best solution. Experience has shown that if the mouth is not less than three feet from the ground and the entry velocity not more than 100 feet per second the susceptibility to FOD during take off and landing is acceptable. The forward surface presented to the airstream should conform to the fuselage lines and be curved in two planes. This will minimise the ice accretion on the nose and centrifuge a large proportion of the water droplets clear of the intake mouth. The intake lip should be semi-circular in section of generous radius and treated with a good thermal insulator to reduce the

catchment efficiency and conductive heat loss. It is essential that an intake of this type is run cold, thus eliminating the problem of runback ice as experienced on heated intakes. Also in the absence of heat, ice which forms within the plenum chamber as a result of droplet separation from the induced flow to the engines will tend to be of a crystaline frangible nature. This type of ice has a very low mechanical strength and is unlikely to cause damage if ingested by the engine. These principles were incorporated in an intake designed for the Wessex 2 and 5. This intake has been used on trials aircraft during the past five winters and has been fitted to a limited number of service aircraft. It has been found to give very good protection in virtually any type of snow over a temperature range of +2°C to -20°C and in light icing. There were, however, occasions when engines were damaged during prolonged periods in relatively heavy icing. This damage resulted from ice building on the rear pressure face and subsequently shedding. The ice in question was of the glime type and probably resulted from operating above the Ludlam Limit where the excess water would tend to produce glaze ice. In an attempt to overcome this problem the trials aircraft was modified by the introduction of a TKS fluid de-icing system designed to dispense fluid on the vulnerable areas. This gave very good results when operating in a classical icing environment, but created problems when operating in snow. In the latter case, snow, which would otherwise be completely innocuous was converted to slush on contact with the de-icing fluid. Slide 5

WINDSCREENS

As the windscreen presents a large flat surface to the airflow it is very vulnerable to ice accretion and can rapidly become obscured in the absence of any protection. Liquid de-icing systems have not been found to be very satisfactory for the following reasons:

a. It is difficult to obtain a good distribution of fluid throughout all flight conditions.

b. Once ice has formed on the screen the application of fluid has not normally removed it.

c. In view of b, continuous operation seems to be necessary which involves a considerable weight penalty.

It is considered that this is one of the few areas where continuous application of heat to prevent the formation of ice appears the best solution. Electrical heating with the aid of a Slide 6

transparent conducting film has been found to be very satisfactory although 28 volt DC systems present some difficulties due to compass interference, and bus bars partially obscuring vision. Problems can however arise due to ice running back from the heated windscreens and suitable precautions must be taken to ensure that this does not become a hazard (eg the fitting of a deflector to divert any shed ice clear of vulnerable areas).

ICE DETECTORS AND OTHER INSTRUMENTS

Mr Abel dealt with this subject in some detail yesterday and I only wish to reiterate the main features of what is considered an unhappy situation:

a. The most satisfactory ice detector tested to date is the 'Hot Rod'. A useful adjunct to this is a modification incorporating a light source and detector which can be adapted to give a visual or audio warning to the pilot. Its main drawback is that it does not give a quick indication of the severity of the encounter. Slide 7

b. No reliable and accurate ($\pm\frac{1}{2}^{\circ}$C) commercial device to measure the outside air temperature in icing conditions has been identified yet.

c. A number of icing severity meters have been used during our trials with some success. However, all of them have some remaining problems which have yet to be solved.

d. A major problem is the siting of the instruments on the helicopter so that they work under all desired conditions of speed from hover to at least cruise speed, altitude and icing and snow conditions.

MAIN ROTOR

It was during the winter of 1960/61 that A&AEE undertook its first major investigation of the effects of icing on a helicopter. The aircraft was a Wessex Mk 1 equipped with an electrically-heated rotor de-icing system. The trial was conducted in the NRC Spray Rig at Ottawa and during the early stages accretion runs were made to establish the optimum thickness of ice on the main rotor blades in order to ensure satisfactory shedding on applying the heat. However, during these runs spontaneous shedding was observed and down to -10°C the shedding was symmetrical and the ice well fragmented. At lower temperatures, the ice accretions tended to become thicker before shedding occurred, the fragmentation was poor and in some instances shedding was asym-

metrical leading to heavy vibration.

For a number of reasons there was a lapse of activity until 1967 when trials were resumed. The aims of the trials were to develop suitable engine protection schemes and to explore the potential of this natural shedding characteristic in relation to a possible limited release to the Services. Preliminary testing in the Spray Rig suggested that some reliance could be placed in natural shedding and the characteristics could be related to the ambient temperature. Firstly there seemed to be a temperature above which no significant ice would form on the rotor blades. This I have referred to as 'the blade icing threshold temperature' and was indicated as approximately $-3^{\circ}C$ for the types tested to date. At temperatures just below this threshold, ice formed in two distinct nodules on the leading edge above and below the stagnation point. Slide 8
At temperatures a few degrees lower the ice formed in a wedge shape over the leading edge with the broad front face of the wedge being concave in the higher temperature range and convex at the lower end of the temperature range. At still lower temperatures the ice formation tended towards rime and lost its Slide 9
wedge shape, so that it was a streamline extension to the aerofoil section. The effect of this ice deposit was to increase the profile drag and hence the power required for any given rotor thrust. It followed that the drag rise for any given rate of deposition was higher at the higher temperatures by the virtue of the shape of ice formation, but for the same reason it was subjected to greater aerodynamic forces tending to dislodge it. The converse applied, and at the lower temperatures the ice formed as a more stable cap. In the case of Sea King and Wessex blades with comparable surface finish (polyurethane protection on the leading edge), the cross-over point was in the region of $-10^{\circ}C$, and down to this cross-over temperature the ice appeared to be shed periodically and symmetrically from the blades at approximately three minute intervals. This I have referred to as 'the critical shedding temperature'. During the accretion there was a progressive increase in the power required for any given flight condition with an associated increase in the blade order vibration. During the shedding which extended over approximately fifteen seconds there was a further marked increase in this vibration which was sometimes accompanied by 1st Rotor order vibration. Immediately following the shed the power and vibration levels returned to near normal, and the cycle repeated indefinitely with very little residual effect. These tests were extended to flight in natural icing conditions, which experience seemed to confirm the finding in the rig leading to expectation of early release for Service operation. During the fourth and fifth seasons of testing there were, however, four isolated occasions when heavy icing was encountered whilst flying in stratus cloud

at temperatures above the 'critical'. On these occasions the natural shedding process appeared to break down and the icing environment was vacated due to excessive torque rise and/or vibration levels. This I consider was probably due to the formation of runback ice on the main rotor blades as a result of operating above the Ludlam Limits. In view of this we renewed our efforts to develop satisfactory in-flight filming techniques in order to obtain photographic evidence of ice accretion and shedding on the main rotor blades. Mr Bradley of our Performance Division did some excellent work in this area. Unfortunately he is unable to be present but he has made the following contribution which I will present for him.

A&AEE has built up a total of 100 hours experience in natural icing amounting to 23 hours (44 flights) with the Sea King, 55 hours, (98 flights) with the Wessex Mks 3 and 5, 9.40 hours (21 flights) with the Puma and 12 hours (19 flights) with Scout and Wasp. I think you will agree that this amounts to a considerable body of well documented experience. What never ceases to surprise us are the variations in aircraft behaviour that can occur from one icing encounter to the next and the consequent difficulty of drawing conclusions with a high degree of confidence even with the relatively large sample of results which we have acquired. To illustrate the variability of icing conditions that can be encountered, the next slide shows two pictures of blade ice acquired by a pair of Wessex helicopters operating at the same time within 15 miles of each other over similar terrain. One aircraft, with blade ice of type (a) was able to sustain flight in icing for a prolonged period while the other made several encounters each one being terminated after a few minutes because of the rapid torque and pitch increases of large magnitude which occurred. Slide 10 & 11

I will now present a time history of what I will boldly describe as a 'typical' flight in natural icing in that the pattern shown might be expected to hold good for some 90% of our excursions into natural icing. Slide 12

Here you see a prolonged flight in icing conditions; from the 'Hot Rod' we deduce a mean rate of about 1 mm minute equivalent to moderate icing conditions. I particularly want you to notice how the torque and pitch increments vary, the maximum torque increment being 30%, at which value the flight can still proceed quite comfortably although of course at the penalty of increased fuel consumption and hence reduced range. The auto rotative checks performed, at intervals, gave fairly small rpm losses and increases in the rate of descent. The rotor head camera film showed ice extending out to at least 80% of the rotor blade

span throughout this flight, except during autorotation when due to the increase in OAT during each descent the ice progressively shed in varying degrees; this probably accounted for the low RPM loss in autorotation.

Unfortunately we have found that on a not insignificant number of occasions the tolerable characteristics shown in the previous slide alter and the rotor power demands can approach values at which level flight can no longer be maintained even at minimum power speed. A time history of such a flight is shown in the next slide. I regret that the information is not so well documented on this occasion. I would point out that this flight was abandoned after 25 minutes when full power produced a rate of climb of only 300 ft/min at the best climbing speed and when an attempt at autorotation resulted in the auto-rpm limit specified for the test being reached at flat pitch with the engines still contributing 10% of their full power output. Unfortunately no rotor head camera shots are available from this flight. Slide 13

On five flights during our last season's testing we encountered conditions where rapid torque and pitch increases of large magnitude occurred. If we define the % power increase as the increment in power needed to sustain level flight at any given weight, speed OAT and altitude in icing divided by the power required in non-icing conditions (expressing this ratio as a percentage) then at around minimum power speed +15 kts we have seen power increases of 100% per minute up to an increment of 70% at which point prudence dictated that the encounter be terminated. Clearly in this sort of rapidly developing situation any carefully briefed plan of testing designed for less exciting circumstances has to be curtailed in the interests of flight safety. Notice on this flight the relatively high temperature of -2°C and the small amount of ice (1 mm) built on the hot rod. Unfortunately, the rate of icing cannot be accurately determined for such a short encounter with our existing instrumentation. Slide 14

It is interesting to note how the rotor rpm losses were barely measurable in the autorotation performed immediately after this high torque rise. A possible explanation is that the ice on the rotor may have been shed during the descent, due perhaps to higher temperatures or to breaking out into clear air with the consequent change in heat balance on the blades. There may be other explanations with different implications on flight safety.

We have very occasionally encountered handling problems during our flights in natural icing. On two occasions flights have been abandoned because of the increased level of 1R vibration that occurred; on both occasions prior to the icing encounter the

vibration levels had been judged higher than normal (but still acceptable) by the pilots. On one of these occasions with our Scout helicopter the pilot had completed an hour and a half sortie in icing during the morning; two hours later after lunch when going for a further flight in what he assumed were very similar conditions he abandoned the encounter after 3 minutes because the level of 1R vibration made instrument flight nearly impossible; the temperature was about -3°C and 2 mm of ice had built on the hot rod; the torque increase was very small in the order of 5%. However, the vibration level remained unacceptable when the rotor blades were free of ice.

On several occasions during flight at speeds within about 15 kts of the normal flight envelope limits we have encountered high loads in the pitch change arms; on one flight when performing a turn the loads increased well beyond the level at which fatigue damage commences and momentary loss of control was experienced indicating the presence of significant blade stall effects. The next slide shows the ice on one blade recorded by the rotor head camera immediately after this incident.

We have had no experience of any blade instabilities that have been recognised as such during flight, but again I must add that we have explored only a limited flight envelope in icing and with the quite large pitch and torque increases we have seen, I must confess that the possibility of some instability occurring does worry me.

We have made specific attempts to promote ice shedding by means of control inputs consisting of:-

a. a rapid lateral stick reversal; and

b. a collective lever 'pump' action.

At no temperature did any identifiable shedding occur even with power increases of up to 40% present.

I am very hesitant to draw any definite conclusions on the subject of rotor icing, but the available evidence points to the following facts.

a. Significant increases in power required can occur over a wide temperature range including very close to freezing. Just how close is difficult to say with the problems of measuring OAT which we have experienced. However, a 30% power increase has been measured when flying at a temperature of -1°C (this tem-

perature value being the best estimate available based on the corrected instrument readings). Rotor head camera films show the spanwise extent of the ice to be at least out to the 90% point in this sort of situation. The power increases can on occasions occur very rapidly and within one or two minutes reach a level at which the aircraft may be forced to descend.

b. Blade pitch increases are usually necessary to maintain a given speed in level flight when high torque rises occur. However, occasionally we have seen torque increases with no significant pitch increase. I assume that this points to a type of ice producing profile drag without affecting the lifting characteristics of the blades.

c. Rotor head camera films have confirmed that the process of self shedding can take place during flight in natural icing conditions but that in some cases the residual deposits that are left may still cause significant aerodynamic deterioration of the blades.

d. There is evidence that high power increments are associated with high rates of ice accretion. However, in some situations the power increases have occurred very rapidly and owing to the failure of the sophisticated ice detectors to function correctly, it has not been possible to record the icing rates over a short period accurately.

e. The effect of ice on the auto rotative characteristics of rotors varies widely. It can on occasions lead to significant rpm losses and rates of descent increase within a very short time. Even in situations where steady state auto rotation is possible there is insufficient evidence at present to say that the entry to autorotation, and the landing without power from a descent at possibly a higher than normal rate allied with lower than normal rpm, will be survivable.

f. Our tests have shown that a period of flight in clear air, or possibly even in cloud without the presence of icing, causes the ice to come off the blades to the point where normal performance levels are nearly restored. Typically, on landing after a natural icing flight, ice has usually shed outboard of 30-40% span. Unfortunately, our information on the time needed for this to occur at various temperatures is incomplete and there is probably a lower limit of temperature below which this does not happen.

Of the 182 natural icing flights we have made, 11 have been terminated prematurely because the crew considered that either

the performance had deteriorated dangerously or vibration had reached an alarming level. On several aircraft types the icing environment also causes an increased risk of engine failure. Consequently we feel that a survivable engine off landing capability must be retained even for twin engined helicopters; the permissible level of rotor degradation to ensure that this capability is retained is not yet known. We are currently trying to formulate some very restricted form of clearance to allow flight in potential icing conditions up to the point where ice is seen to form followed by an immediate evacuation of the environment. Such a release is very difficult to frame in sensible terms.

TAIL ROTOR

The tail rotor blades are subject to accretion and shedding similar to the main rotor but experience to date has shown that the shedding is superior, and so far no problems have been experienced. This is probably due to the smaller blade section giving rise to higher shear stress at the ice/blade interface and the increased CF gradient across the span of the blade producing higher internal stresses in the ice. Only one tail rotor strike has been found in all our seasons of testing; the damage was negligible. Slide 15

FLYING CONTROLS

The rotating parts of the flying control system, although subject to ice accretion, do not normally suffer any adverse effects as they are in continuous cyclic motion. Any parts of the system subject to random linear movement (eg flying control jacks) if found to be vulnerable can normally be protected by deflectors, which, if properly designed, should not require de-icing. Slide 16

AIRFRAME

Ice deposited on the airframe is unlikely to present any problem from weight growth, as it is mainly limited to those parts which present a relatively small projected area to the airstream. The main problems arise from the build up of ice on external struts, pipes and other projections of small section which have a high catchment rate. In these cases the ice tends to build in a form which will eventually be shed by aerodynamic forces when it extends beyond a certain amount or the shedding may be initiated if the aircraft, having encountered icing conditions, then descends through warmer air. These problems can be reduced by ensuring whenever possible, radio aerials, probes, and other

potential ice catching devices are sited where the ice will shed clear of the vulnerable areas (eg engine intakes, tail rotors, etc). In the case of pipes, struts, etc, which cannot be easily repositioned, the hazard can be reduced by increasing their section, if necessary by the addition of fairings.

CONCLUSIONS

Our experiences during last winter's trials have left us in some doubt about the extent to which the Services will be able to operate helicopters with unprotected rotors in an icing environment. There are clearly limits when the natural shedding characteristics will be swamped giving large increases in torque and possibly vibration. However, we have not yet reached the stage in digesting the data gathered and subsequent analytical studies and supporting tests where we can conclude that reliance cannot be placed in natural shedding within some limited operational envelope.

This is not to say that a rotor de-icing system will solve the problem. Limited experience with electrically heated rotor blades both in early 1960s and in the past two winters have shown that these systems also can be saturated in heavy icing conditions. Furthermore the cases where large torque rises have occurred with very limited exposures and presumed relatively light icing on the blades, suggest that the postulate of allowing ice to accrete to a given thickness (usually 6-8 mm) before initiation of shed may be invalidated.

Nevertheless, if some Service operation of helicopters is deemed possible we are currently thinking of framing the release in three categories:

i NORMAL FLYING Forecast Icing, which will permit cloud penetration, but the environment must be vacated when ice starts to form on the airframe. Virtually unrestricted flight out of cloud in all forms of precipitation excluding Freezing Rain.

ii FLYING TRAINING This should permit deliberate continuous flight in icing conditions subject to maintaining an escape route, in terms of weather minima, which would enable the pilot to descend out of cloud and regain good visual contact.

iii OPERATIONAL NECESSITY In this category the decision to launch or continue the mission would rest with the authorising officer and captain of the aircraft. This would normally involve flying in conditions from which there is no

immediate escape, with an associated higher risk.

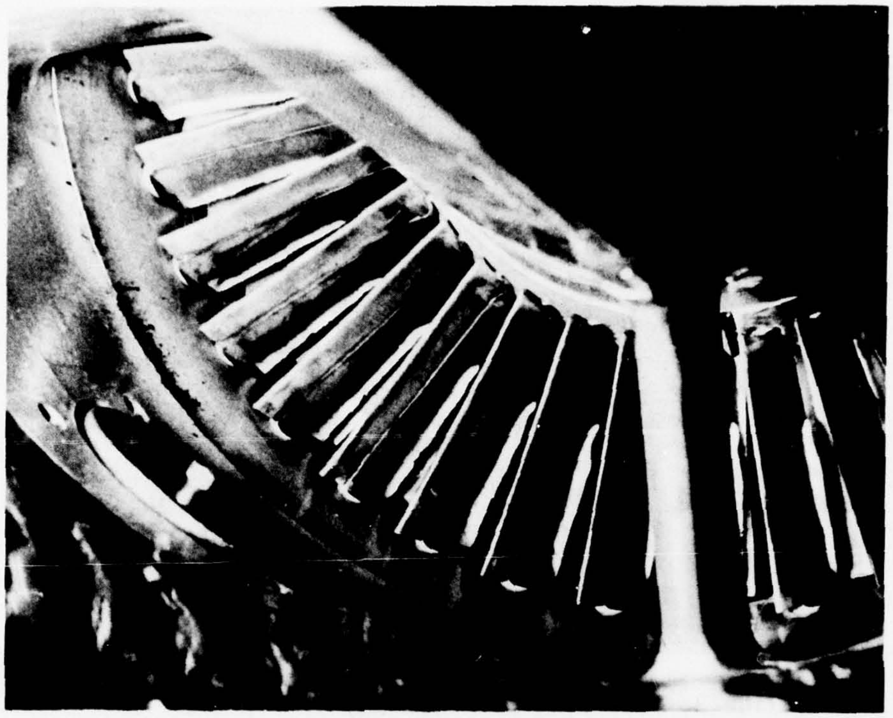
In order to exploit the training release fully and to minimise the risk in the case of operational necessity, the pilot must be fully informed on:

a. The limitations of his aircraft.

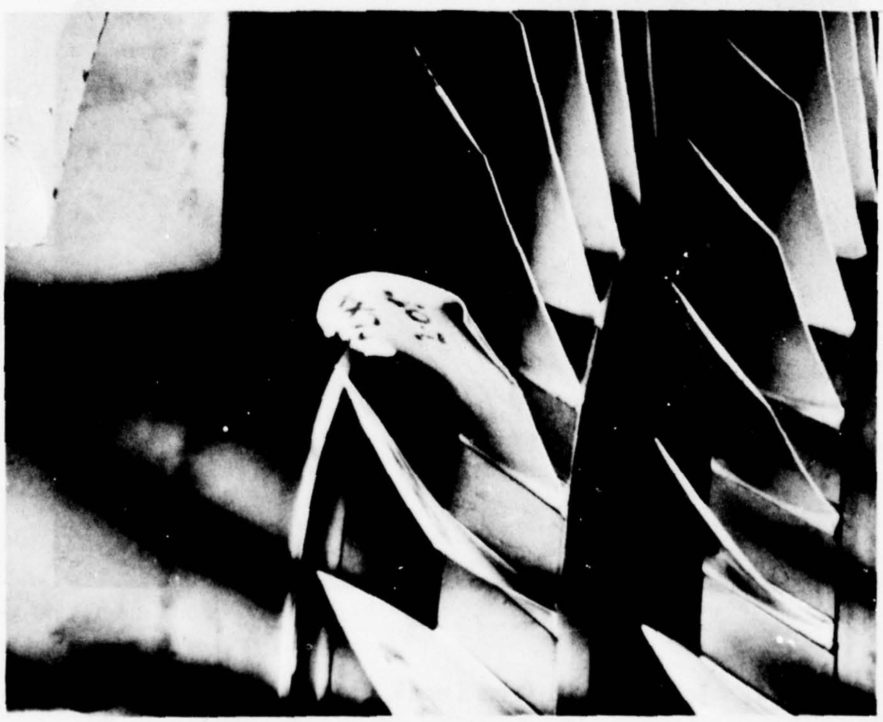
b. The reasons for the limitations.

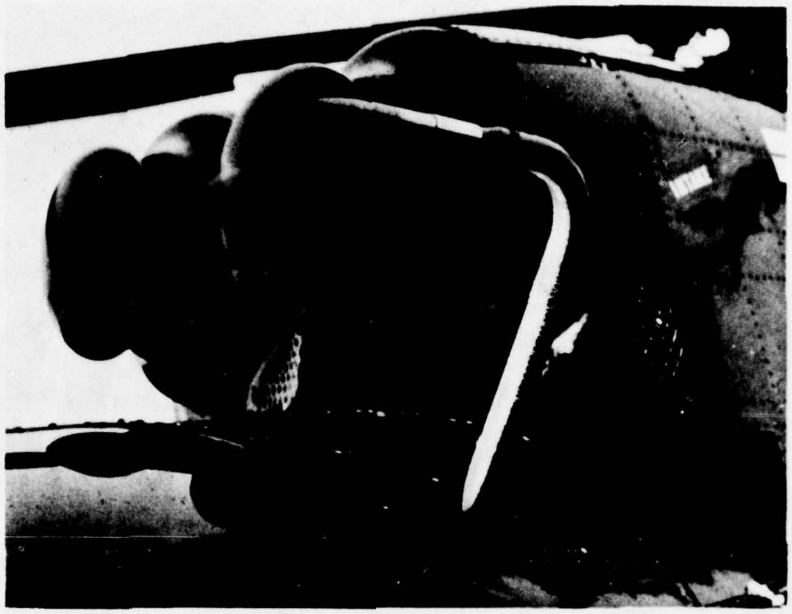
c. The maximum use of the instruments available to him in order to identify trends before the situation gets out of hand.

Finally, I would like to say that in the field of helicopter icing, I doubt if we shall ever be in a position to 'Rest on our arms reversed' and say we have achieved an entirely satisfactory standard of safety. One must accept an added risk and there are likely to be occasions where immediate exit from the icing conditions will be the only safe action to take. However, improvements in meteorological forecasting, ice detection etc should eventually minimise the number of such occasions.

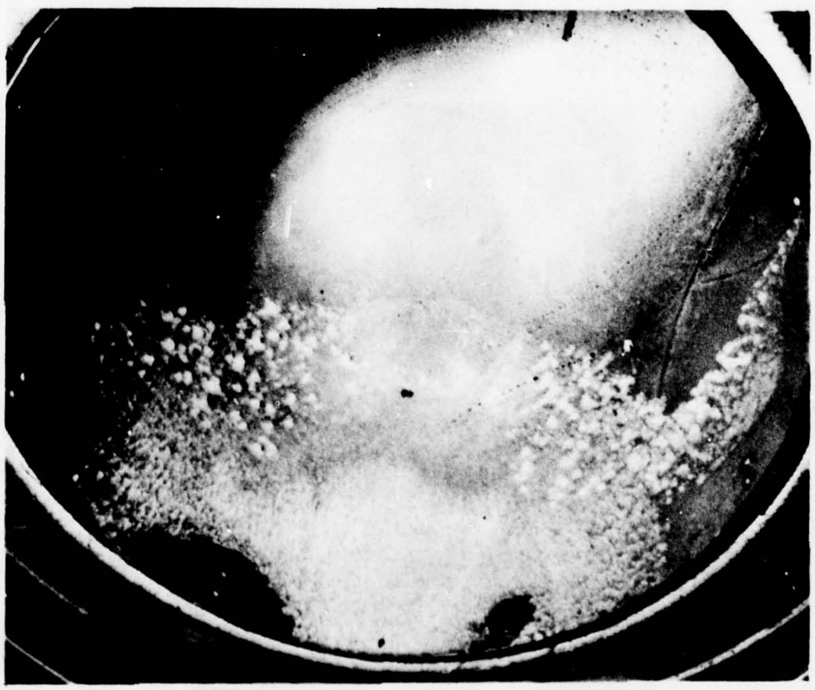


Slide 1 & 2. Typical Engine Ice Ingestion Damage.





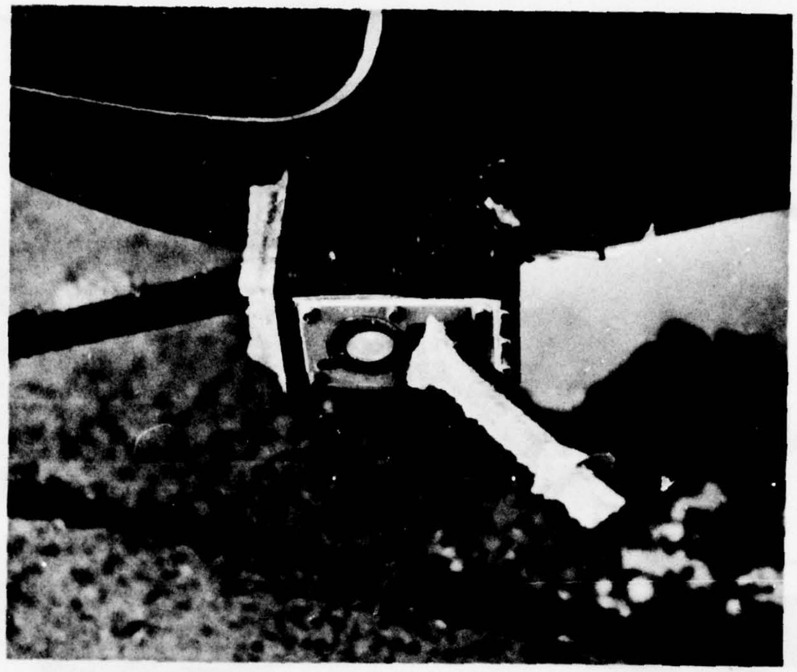
Slide 3. Sea King Mushroom Engine Intake.



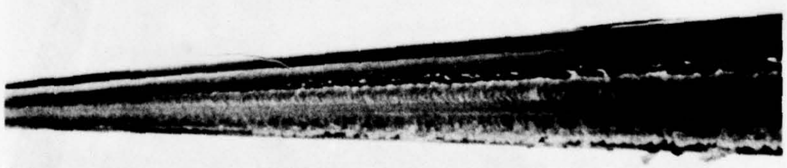
Slide 4. Ice Deposit from Local High Impingement.

Slide 5. Downward-facing Plenum Intake.

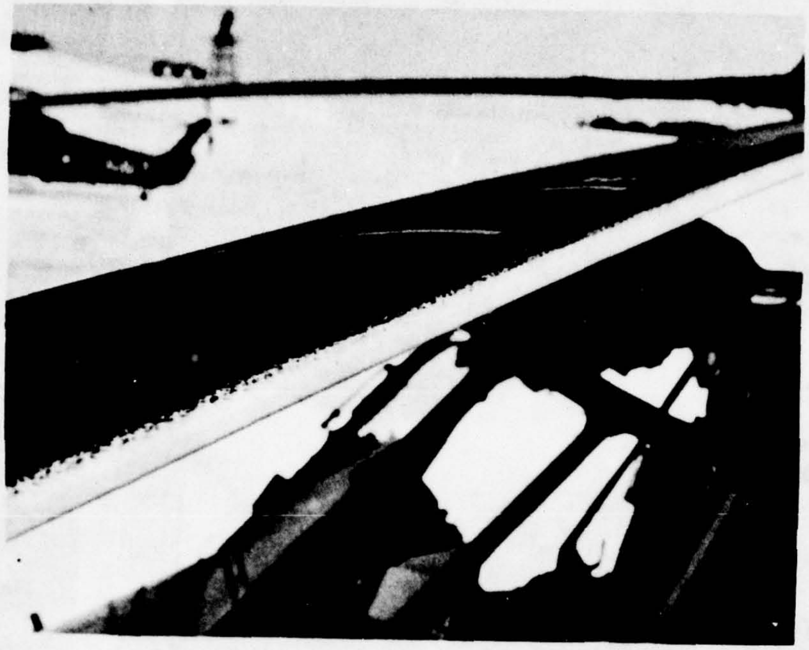
Slide 6. Scout Electrically Heated Windscreen.



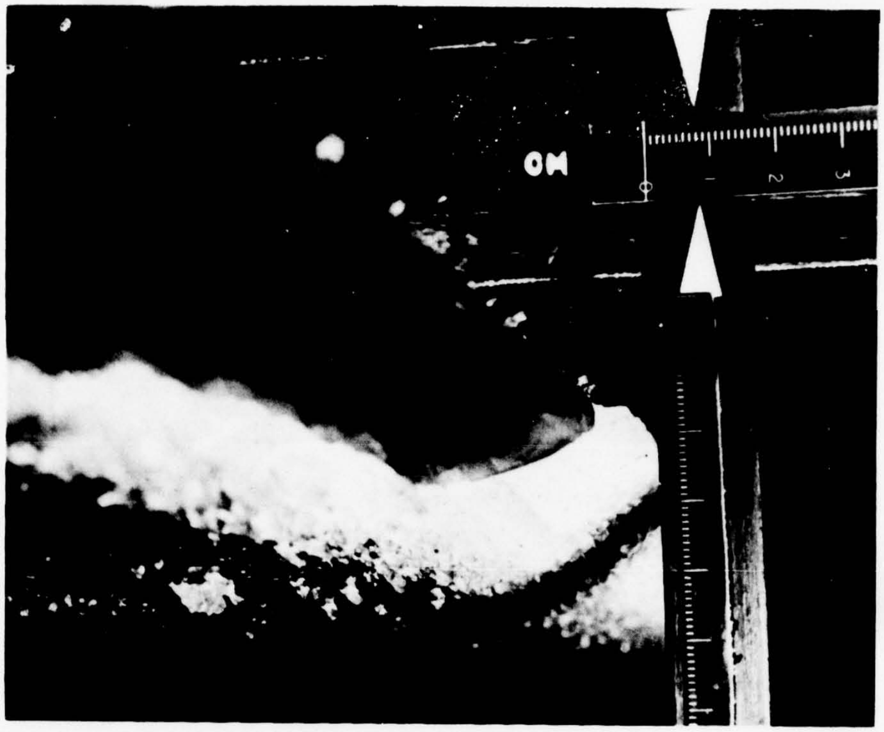
Slide 7. "Hot Rod" Ice Detector.



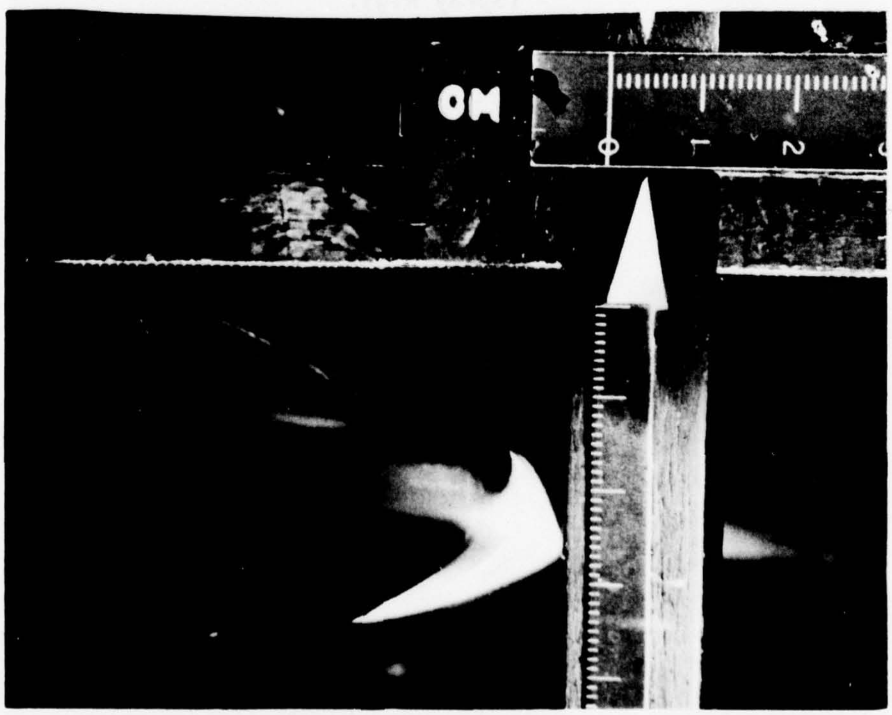
Slide 8. Typical Ice Accretion on M.R. Blade at -4^{o}C. (Spray Rig).



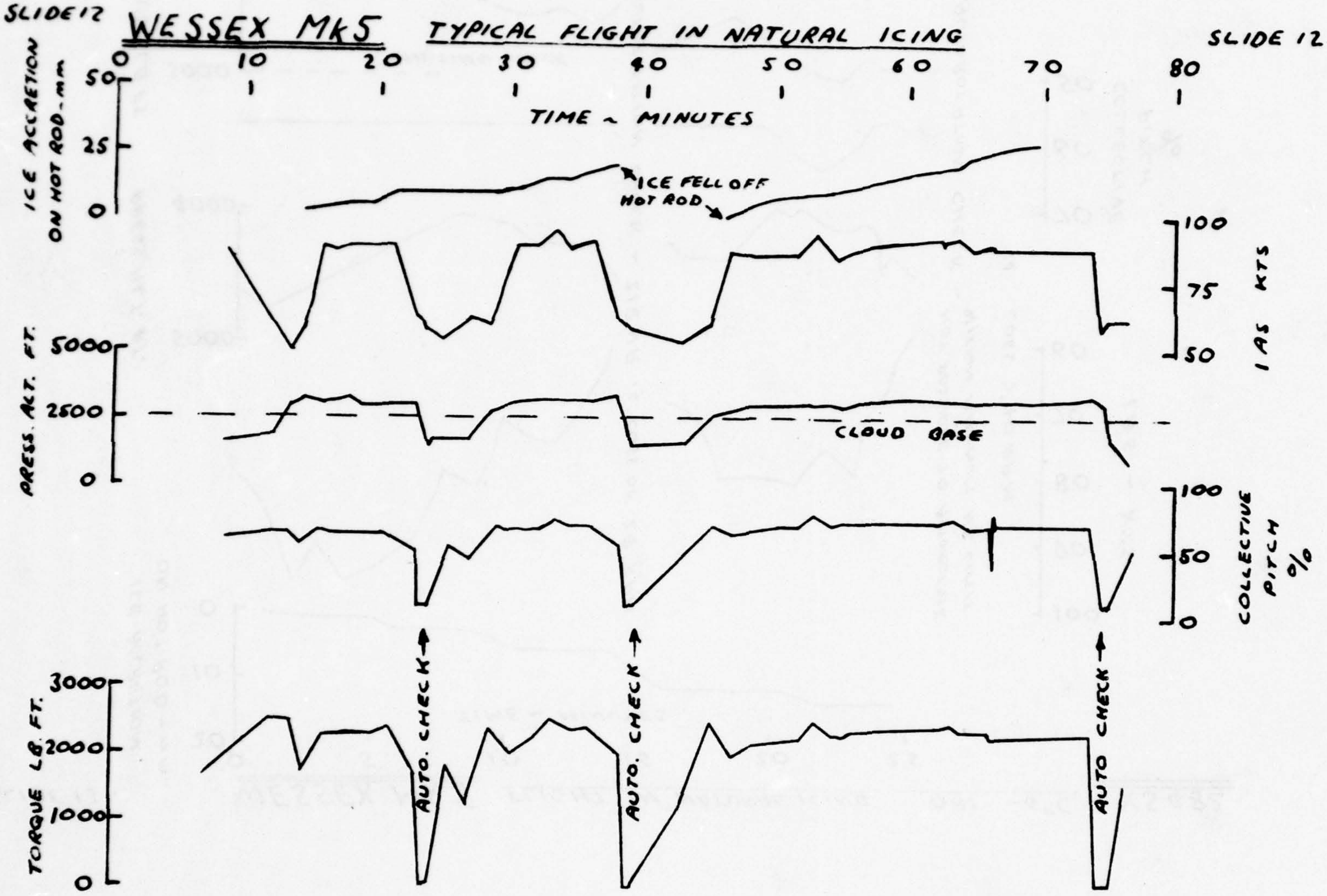
Slide 9. Typical Ice Accretion on M.R. Blade at -10^{o}C.
(Spray Rig).



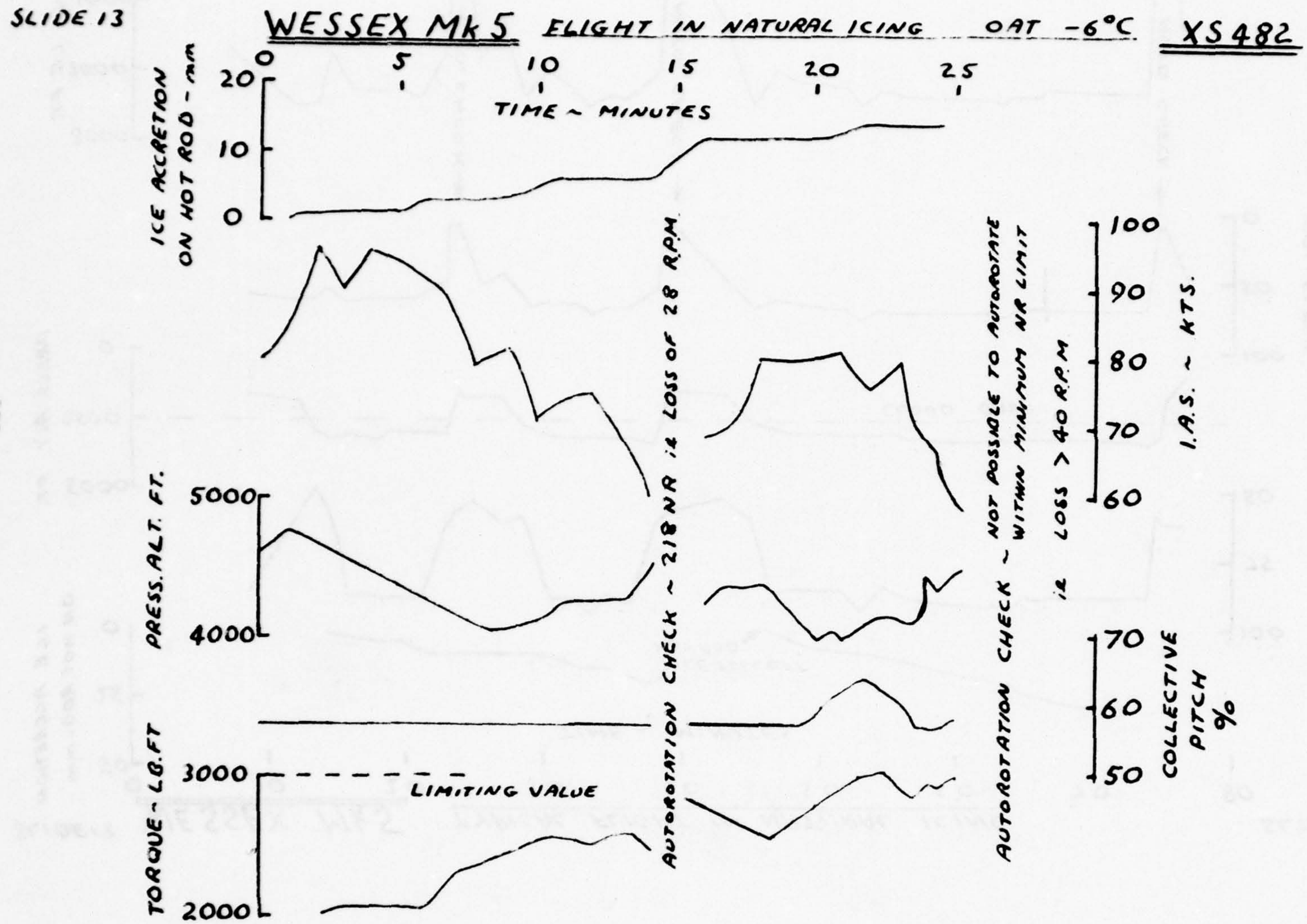
Slides 10 & 11. Comparison of M.R. Blade Ice acquired by a pair of Wessex Helicopters operating at the same time within 15 miles of each other.



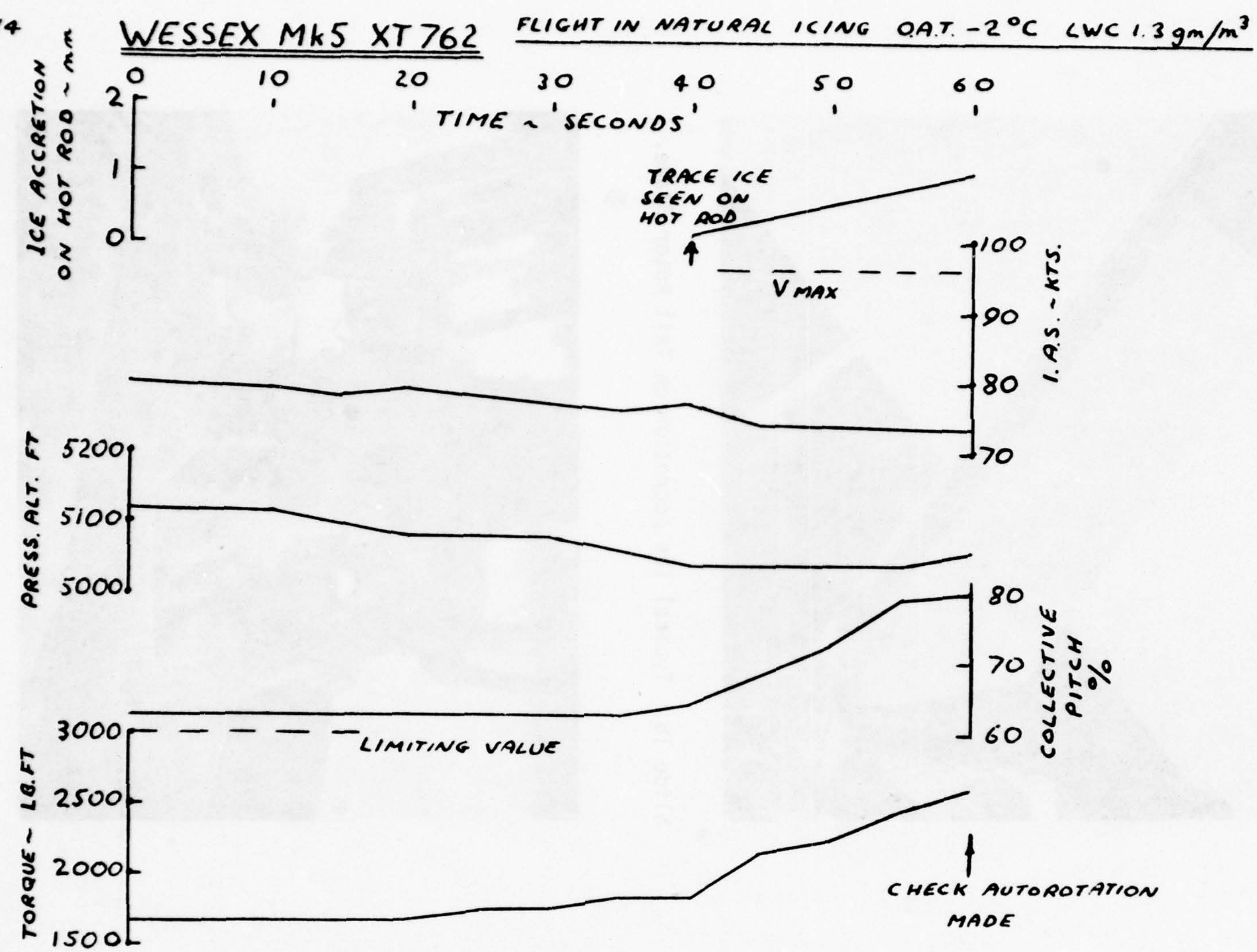
SLIDE 12



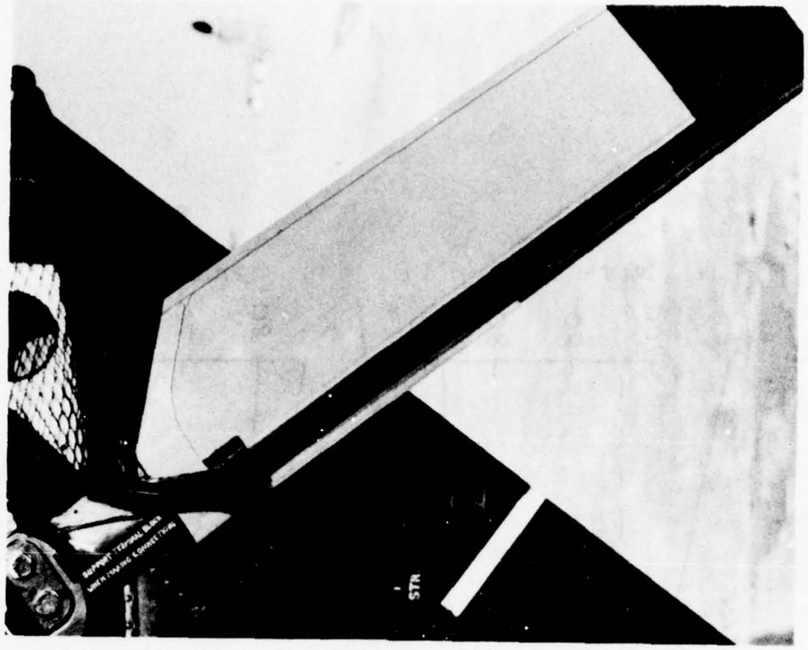
WESSEX MK5 TYPICAL FLIGHT IN NATURAL ICING
SLIDE 12
0
10
20
30
40
50
60
70
80
TIME ~ MINUTES
ICE ACCRETION ON HOT ROD - mm
50
25
0
ICE FELL OFF HOT ROD
100
75
50
IAS KTS
PRESS. ALT. FT.
5000
2500
0
CLOUD BASE
100
50
0
COLLECTIVE PITCH %
AUTO. CHECK
AUTO. CHECK
AUTO CHECK
TORQUE LB. FT.
3000
2000
1000
0


SLIDE 13
WESSEX Mk 5 FLIGHT IN NATURAL ICING OAT -6°C XS 482
TIME ~ MINUTES
0
5
10
15
20
25
ICE ACCRETION ON HOT ROD - mm
20
10
0
PRESS. ALT. FT.
5000
4000
TORQUE - LB.FT
3000
2000
LIMITING VALUE
AUTOROTATION CHECK ~ 218 NR i.e. LOSS OF 28 R.P.M.
AUTOROTATION CHECK ~ NOT POSSIBLE TO AUTOROTATE WITHIN MINIMUM NR LIMIT i.e. LOSS > 40 R.P.M.
I.A.S. ~ KTS.
100
90
80
70
60
COLLECTIVE PITCH %
70
60
50

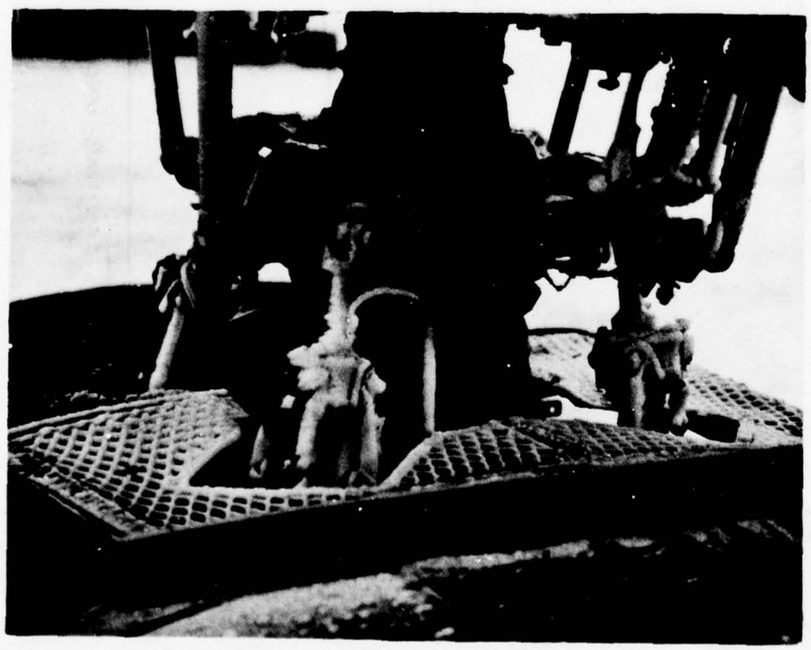
SLIDE 14



WESSEX Mk5 XT762
FLIGHT IN NATURAL ICING O.A.T. -2°C LWC 1.3 gm/m³
TIME ~ SECONDS
0
10
20
30
40
50
60
ICE ACCRETION ON HOT ROD ~ mm
2
1
0
TRACE ICE SEEN ON HOT ROD
V MAX
100
90
80
70
I.A.S. ~ KTS.
PRESS. ALT. FT
5200
5100
5000
80
70
60
COLLECTIVE PITCH %
LIMITING VALUE
TORQUE ~ LB.FT
3000
2500
2000
1500
CHECK AUTOROTATION MADE



Slide 15. Typical Ice Accretion on Tail Rotor Blade.



Slide 16. Ice Accretion on Wessex Flying Control Jacks.

THE ABILITY OF THE CLIMATIC LABORATORY TO PRODUCE A WIDE VARIETY OF CLIMATIC CONDITIONS

Richard Toliver
Icing Research Engineer

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32542

Introduction and History

In 1934 the Baker Board recommended that tactical units be trained under various environmental conditions. This recommendation and the interest of General H. H. Arnold led to the establishment of Ladd Field at Fairbanks, Alaska in 1942 and cold weather testing was begun on a regular basis.

Ladd Field faced many problems, the major problem being that weather, because of its uncertainty, played havoc with the schedules. This problem, plus the fact that much of the Army Air Force equipment at this time could not be used in temperatures below -20°F, made it clear to the Army Air Force that a more positive means of cold weather testing must be found.

The Army Air Force cold weather testing mission was then assigned to the Armament Development and Test Center and primarily through the efforts of Colonel H. A. Russell and Lt Colonel A. C. McKinley, plans were completed for building the Climatic Laboratory. The construction cost of the Laboratory when completed was 5.5 million dollars. Replacement cost today is estimated to be 50 million dollars.

The first major cold test in the facility was in May of 1947. Systems tested at this time included the B-29, P51, P35, and R5D helicopter. These early tests were so successful and yielded so much useful data that the Laboratory was soon made available to all Government agencies and sponsored contractors.

Mission of the Climatic Laboratory

The mission of the Climatic Laboratory is to provide global climatic environmental testing conditions in order that the United States Air Force and other agencies of the Department of Defense may develop and test weapons and weapon support systems as required by Air Force Regulation 80-31 and other applicable directives. The Laboratory has supported development and testing programs for many agencies including Air Force, Navy, Coast Guard, Army, Marine Corps, NASA, National Weather Service,

National Science Foundation, ARFA, AFCRL and the Tennessee Valley Authority.

The basic environmental test effort at the Laboratory is divided into two areas: (1) basic research and development work associated with prototype items or new systems, and (2) operational services associated with particular climatic related problems of in use equipment.

In order to accomplish the mission of the Climatic Laboratory, the assigned personnel provide a variety of services including:

(1) Operate and maintain eleven testing facilities to provide various environments for military testing under approved programs.

(2) Provide consultant engineering services for users of the facilities.

(3) Design, manufacture, and install test support equipment.

(4) Design, fabricate and install instrumentation and data collecting systems.

(5) Provide liaison and monitoring services for testing teams.

(6) Manufacture test item fixtures/hardware as required.

(7) Accomplishes hardware/system failure analyses.

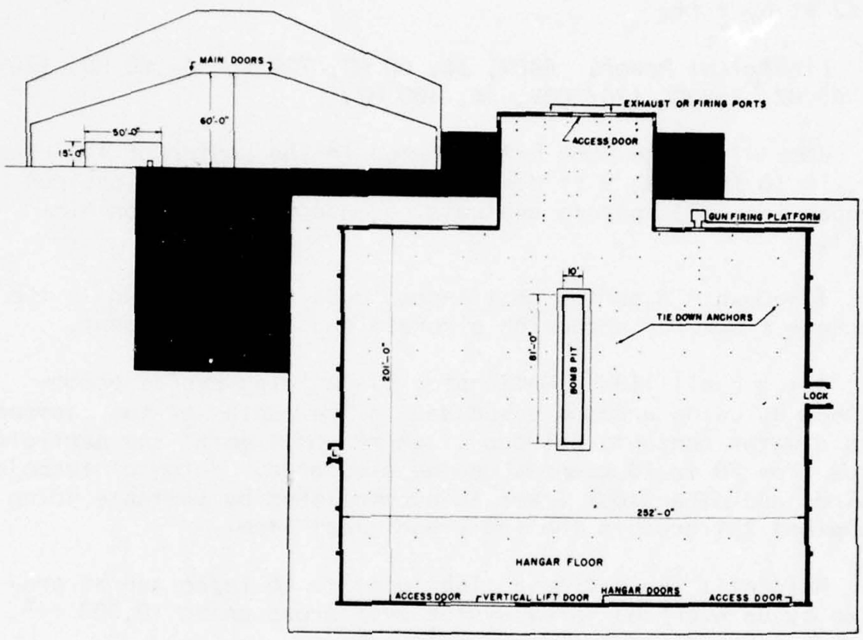
Test Chambers - Their Size and Capabilities

In the eleven test chambers of the Climatic Laboratory, conditions can be simulated from the unbearable heat and sunshine of a desert to the bitter cold and gale force winds of the Arctic. Temperatures from 82°C to -75°C and winds up to 145 knots can be simulated.

Main Chamber

The Main Chamber (MC) of the Laboratory is unique in that it is the largest and most complex climatic environmental test chamber in the world. The chamber is an insulated hangar having a total enclosed volume of 3,282,500 ft^3. The size of the

chamber permits testing of very large items of equipment and complex weapon systems. Several tests can be conducted simultaneously, depending upon the size and complexity of the individual items. Detailed specifications of the MC is as follows:



Size: 252 ft wide, 201 ft deep, 70 ft high in center, 35 ft high at sides with an appendant floor area 60 ft by 85 ft and a ceiling height of 75 ft. Usable floor space is 55,000 sq. ft.

Temperature Range: 76°C to -55°C. Maximum chamber cooling rate from +20°C to -55°C in 24 hours.

Humidity Control: From 10% to 95% within $\pm 5\%$.

Air Make-Up: To maintain precise temperature control during jet engine runs, up to 650 #/sec of air at temperatures as low as -55°C can be ingested into the chamber to replace air that is heated and exhausted out of the chamber by the jet engines.

Access: Main Doors - 250 ft wide, 60 ft maximum height, opening time is 8 minutes at -65°C. Vertical Lift Door (within the main doors) 15 ft high, 50 ft wide, operating time is 2 minutes. South Lock Door - 10 ft by 10 ft. East Wall Door - 14 ft by 14 ft (gives access to the rear of the hangar through the appendant floor area.) Eight personnel doors - approximately 3-1/2 ft by 7 ft.

Electrical Power: 440V, 3ϕ, 60 HZ; 220 V, 3ϕ, 60 HZ; 120V, 1ϕ, 60 HZ, 28VDC, 120/208V, 3ϕ, 400 HZ.

Bomb Pit: The Bomb Pit, located in the center of the chamber, is 10 ft wide, 9 ft deep, and 80 ft long. Munitions can be dropped into this pit to evaluate release mechanisms on aircraft.

Tie-Down: A series of tie-down eyes are installed in the concrete floor for anchoring aircraft and other equipment.

Icing Facilities: Helicopter blade icing can be accomplished by using a frame suspended in the center of the chamber. Liquid water contents between .1 gm/m^3 and 4 gm/m^3 and particle sizes from 20 to 70 microns can be simulated. Icing of turbojet engines and other test items is accomplished by portable icing equipment tailored to the individual test item.

Rainfall: Rain from a light mist to 15 in/hr can be produced by an overhead spray system over areas up to 10,000 ft^2.

Instrumentation: The chamber is provided with three instrumentation terminals each having 192 pairs of number 16 shielded instrumentation lines. One each terminal is located on the south, east and north walls of the chamber. The lines are terminated in a patch panel located in the data recording room. A total of 400 data channels can be recorded at one time.

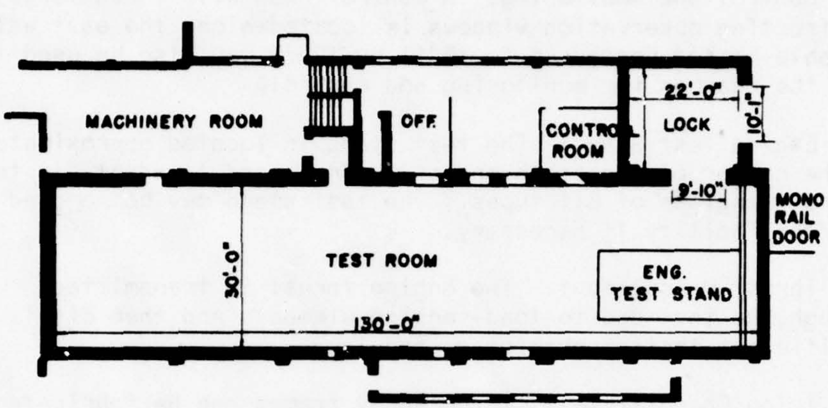
Gun Firing: A projectile trap permits gun firing of automatic weapons up to 30mm. Request for gun firing support is evaluated for safety based upon rate of fire, velocity, caliber and length of burst.

Snow: Snow can be made in the main chamber in varying densities and up to 200 ft^3 per hour.

Control and Monitoring: An Observation Room with three large non-frosting observation windows is located on the south wall and a similar room on the north side of the chamber.

Portable heated booths up to 12 ft X 12 ft may be used inside the chamber for control and monitoring.

Engine Test Facility



The Engine Test Facility is used primarily for environmental qualification and external and internal icing of turbojet engines. This facility is supplied with refrigeration from the same system that supplies the Main Chamber. Therefore, due to its smaller size, the temperature range is greater (+77°C to -77°C) and the maximum cooling rate is much faster (15°C to -62°C in 7-1/2 hrs). Permanently installed instrumentation includes equipment for recording temperatures from -75°C to 165°C, gauge lines for recording pressure at over 100 pickup points, and a vibration indicating system. A high speed Digital Data Acquisition System can be used to write computer input tapes. In addition, oscillographs are available for recording transient data, and a closed circuit television network can be used to facilitate testing. Specific details of the facility are:

Size: 130 ft long, 30 ft wide, 22 ft high. Usable floor space is limited to approximately 100 ft by 30 ft because of two 1000 gallon fuel storage tanks located within the room to facilitate fuel cold/hot soak for engine testing.

Air Make-Up: Same as Main Chamber.

Access to Facility: An electrically operated door, 30 ft by 25 ft opens the entire south wall of the room. An air lock,

22 ft square, is provided in the southeast corner of the room. Two personnel doors, 3-1/2 ft by 7 ft, are provided in the east wall.

Control and Monitoring: A control room with three large, non-frosting observation windows is located along the east wall. Portable heated booths up to 12 ft by 12 ft may also be used inside the chamber for monitoring and control.

Engine Test Stand: The test stand is located approximately in the center of the south end of the room and is adaptable to aircraft engines of all types. The test stand may be removed from the facility if necessary.

Thrust Measurement: The engine thrust is transmitted through the test bed to load-sensing elements and then displayed on different instrumentation as required.

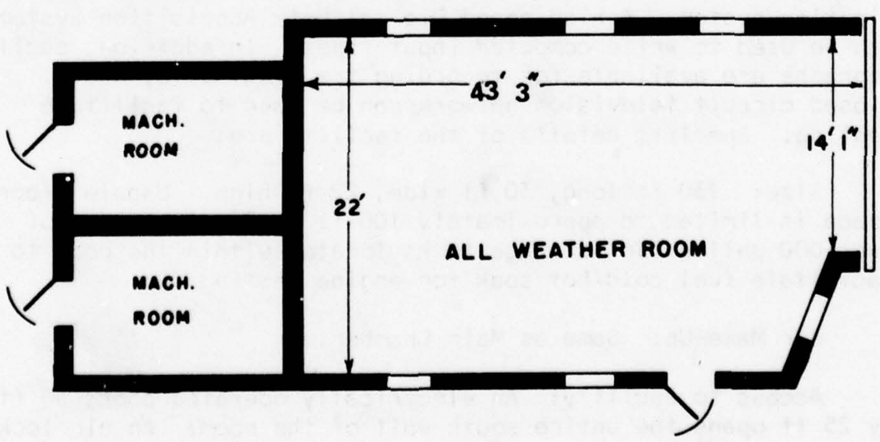
Icing Capabilities: Icing spray frames can be fabricated to meet the particular requirement for engine icing. Fuel icing is accomplished by preconditioning the fuel, measurement of liquid water content of the fuel, and temperatures control of the fuel and the test space to produce icing conditions.

Humidity: Same as Main Chamber.

Electrical Power: Same as Main Chamber.

Snow: Same as Main Chamber

<u>All Weather Room</u>



One of the most versatile facilities in the Climatic Laboratory is the All Weather Room. In this chamber, arctic to jungle conditions can be produced on an overnight basis. Rainstorms up to 12 inches of rainfall per hour can be created. Sand and dust storms as well as snow can be produced as the test requirements demand. Its specifications are:

Size: 43 ft 3 in by 22 ft by 15 ft 7 in.

Temperature Range: +77°C to -62°C Maximum cooling rate is from +15°C to -62°C in 24 hours.

Humidity Control: From 5% to 95% with regulation to within ±2%.

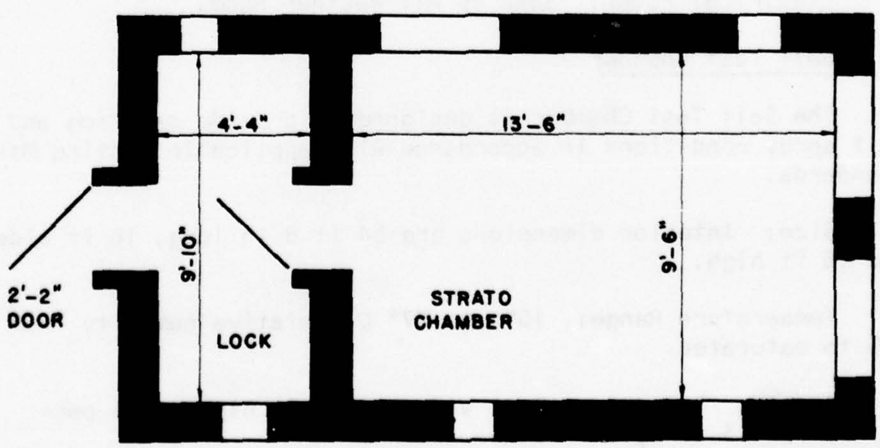
Air Make-Up: Maximum of 500 cfm.

Access: A vertical lift door 14 ft wide by 15 ft 6 in high opens to an outdoor ramp on the east end of the room. A personnel door 2-1/2 ft by 6 ft 6 in is provided on the south wall. Small doors, ports and windows are located within the walls for viewing and access to instrumentation and power leads.

Sand and Dust: Sand and dust are introduced into the chamber from outside sources and are maintained in suspension by wind machines.

Electrical Power: 440V, 3ϕ, 60 HZ, 120/208V, 3ϕ, 60 HZ; 120/208V, 3ϕ, 400 HZ, 28 VDC.

Physiological Strato Chamber



The Physiological Strato Chamber is the Laboratory's only chamber in which altitude as well as temperature can be simulated. Temperatures to -70°C and simulated altitudes as high as 80,000 ft (20mm Hg) have been attained. With precooling, the temperature in this chamber can be reduced from +20°C to -56°C and at the same time pressure can be reached to 87mm Hg (50,000 ft), in 12 minutes.

A lock is provided so that personnel may enter and leave the chamber with minimum interference to test conditions. In addition, instrumentation panels are provided to facilitate the recording and control of test data.

The chamber is constructed of welded steel, insulated with 13 sheets of reflective metal insulation in the chamber and 7 sheets in the lock. The chamber is constructed to withstand pressures from zero absolute to one atmosphere.

Size: The chamber is 13-1/2 ft long, 9-1/2 ft wide, and 6 ft 10 in high. A lock adjacent to the chamber is 9 ft 10 in wide and 4-1/3 ft long.

Temperature Range: +60°C to -70°C

Communications: An aircraft-type intercommunication system is provided for communication between the outside and the chamber. Six observation windows are provided.

Access: Door size for both the lock and chamber is 2 ft 2 in by 5 ft 8 in.

Electrical Power: Same as All Weather Room.

<u>Salt Test Chamber</u>

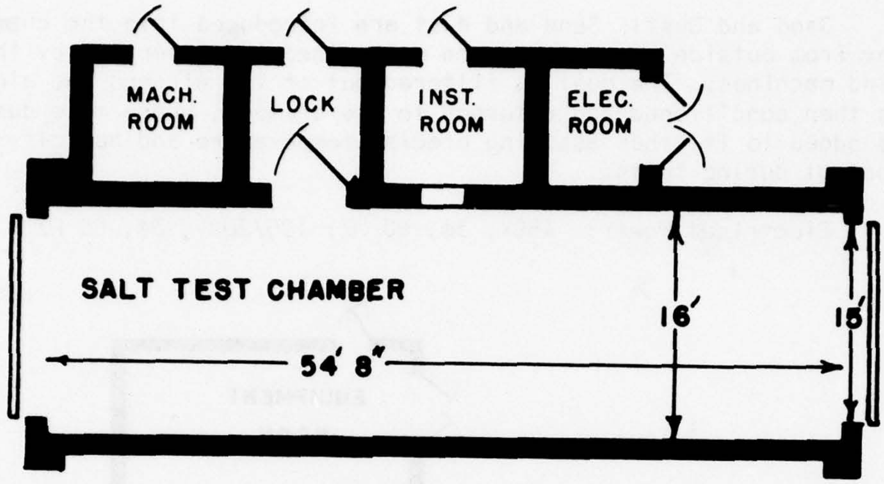
The Salt Test Chamber is designed to provide salt fog and salt spray conditions in accordance with applicable testing MIL standards.

Size: Interior dimensions are 54 ft 8 in long, 16 ft wide and 16 ft high.

Temperature Range: 10°C to 77° C, relative humidity from 30% to saturated.

Access: End doors 15 ft wide and 15 ft high - one personnel door 3 ft by 6 ft 6 in.

Electrical Power: Same as All Weather Room



Salt Test Chamber

Sun, Wind and Rain Room

The Sun, Wind and Rain Room is designed to facilitate MIL-STD-810B standard rain, solar radiation, and dust testing. Rainfall from 1 in/hr to 12 in/hr can be simulated over the entire 50 ft by 50 ft floor of the chamber. Winds up to 40 mph can be simulated and these winds can be in conjunction with the rain or the sand and dust.

Size: The inside dimension of the chamber is 50 ft by 50 ft by 25 ft.

Temperature Ranges: 21°C to 70°C.

Humidity Control: From 95% at 71°C to 20% at 20°C.

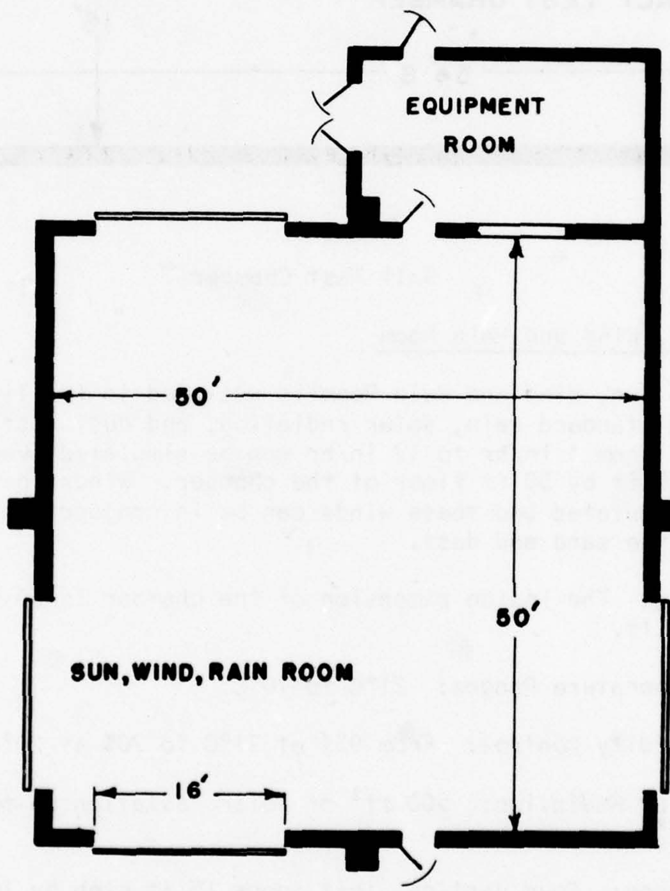
Solar Radiation: 500 ft^2 of solar radiation up to 120 $watts/ft^2$.

Access: Four vertical left doors 15 ft high by 16 ft wide, are provided in the chamber. The room also has two personnel doors. An observation window and several ports are located in the control room for viewing and access to instrumentation.

Wind Machines: Up to three wind machines can be used at one time. Each machine can produce velocities from 5 to 35 knots.

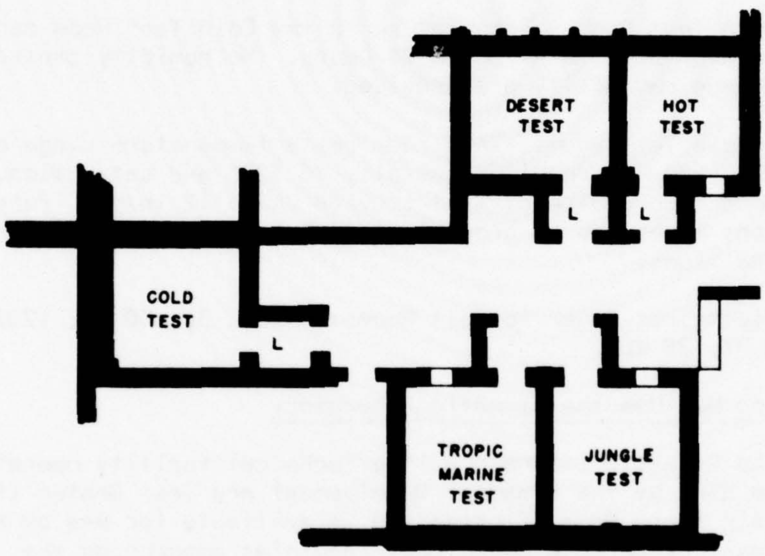
Sand and Dust: Sand and dust are introduced into the chamber from outside sources and are maintained in suspension by the wind machines. The dust is filtered out of the air and the air is then conditioned and returned to the chamber, where more dust is added to it, thus assuring precise temperature and humidity control during tests.

Electrical Power: 480V, 3ϕ, 60 HZ; 120/208V, 3ϕ, 60 HZ



Small Test Rooms

The five small test rooms complete the global environmental capability of the Climatic Laboratory. Within the complex formed by these rooms, it is possible to travel from arctic to jungle to desert to tropical environment in a few steps. These facilities are used to conduct individual tests as well as to serve as aids to testing in the larger chambers, thereby avoiding delays during the major test periods. The rooms included in this complex are the Desert Room, the Hot Test Room, the Tropical Marine Room, Cold Test Room, and the Jungle Test Room.



Small Test Rooms

Each Small Test Room is approximately 12 ft by 12 ft with a ceiling height of 9 ft. All rooms are provided with 3 ft by 7 ft doors and non-frosting observation windows.

Desert Room: A bank of 144 sunlamps on 10 in centers produce an approximation of the noon day desert solar radiation. Room conditions may be varied from +15°C and 40% relative humidity to 40°C and 7% relative humidity. Temperature and relative humidity are controllable to within ± 2°C and ± 2% respectively and test conditions can be maintained for an indefinite period.

Hot Test Room: The temperature range of the Hot Test Room is 21°C to 77°C with relative humidities between 10% and 90%. Temperature and relative humidity are controllable to within ±1°C or ±1% respectively, and may be stabilized for an indefinite period.

Tropical Marine Room: Conditions in this room may be cycled within three hours from 40°C and 50% relative humidity to 21°C and saturation. Rain producing equipment will provide up to 12 in/hr. Salt fog can be introduced into the chamber and salinity of the water can be controlled to duplicate that of any area of the globe.

Cold Test Room: Temperature in the Cold Test Room can be reduced from 32°C to -57°C in 24 hours. No humidity control or air make-up capability are provided.

Jungle Test Room: This room has a temperature range of from 43°C and 80% relative humidity to 32°C and saturation. Rain producing equipment will provide up to 12 in/hr. Fungi of different types can be grown in this from under typical tropical conditions.

Electrical Power for all Rooms: 440V, 3ϕ, 60 HZ; 120/208V, 3ϕ, 60 HZ, 28 VDC.

Who May Use the Climatic Laboratory

The Climatic Laboratory is a technical facility operated for the USAF by the Armament Development and Test Center (ADTC), Eglin Air Force Base, Florida and is available for use by all Department of Defense agencies. Inquiries concerning the technical capabilities of the Laboratory should be addressed to ADTC (TSW), Eglin AFB, Fl 32542; general program test negotiations should be conducted with ADTC (TGPS), Eglin AFB, Fl 32542.

Testing periods for the Main Chamber must be scheduled well in advance, however, they are flexible where possible in order to provide maximum chamber utilization. Testing periods for the smaller auxiliary chambers are arranged according to the specific requirements of the testing agency.

It is suggested that, whenever practicable, an advance planning meeting be held at ADTC, with representatives from the sponsoring agency and the testing agency or contractor in attendance. This meeting should be arranged through ADTC (TGPS)

and should be planned with the following purposes in mind:

a. Afford all test personnel the opportunity to review, define, clarify, and reach agreement on the test support requirements. It should be possible at this meeting to identify those requirements which can be satisfied by resources at ADTC, those which will require procurement action by ADTC, and those which the sponsoring agency may be required to provide.

b. Provide prospective users the opportunity to become familiar with the details of operation and the capabilities of the Laboratory.

Requests for testing involving use of the Main Chamber should be forwarded to ADTC (TGPS), 90 days in advance, if possible, to allow time for planning purposes. A lead time of 30 days is desirable for tests to be conducted in any of the smaller chambers.

No action can be taken on requests for tests which are received directly from industry; however, contractors holding contracts with any agency of the Department of Defense may conduct tests in the facility when sponsored by the DoD agency with whom they are under contract.

Under certain conditions, commercial non-government contractors desiring the use of the Climatic Laboratory facilities may be sponsored by a Federal executive agency (non-DoD). These conditions are outlined in Air Force Regulation 80-19.

Upon receipt of a test request from the DoD testing agency or the DoD sponsoring agency, ADTC will review the support requirements and will advise all concerned of ADTC acceptance or rejection, and of any recommended changes to the test procedures or support requirements. No test, however, will be supported without an official letter or message from the responsible DoD agency requesting support of the test and approving expenditure of necessary funds. After receiving an acceptance notice from ADTC, the requesting agency must notify ADTC of its firm intent to conduct the test. Upon receipt of this notice, ADTC will schedule the test.

For emergency tests, ADTC will immediately review each request for test support on its own merits, and will consider providing support on an "as available" basis, making every effort to cooperate with the responsible DoD agency.

Conclusion

Out of the years of testing experience has grown a philosophy equally applicable to all services and agencies of the Department of Defense. This philosophy embraces the concept that environmental testing of all systems is a vital necessity in the improvement of overall system reliability. Through both simulated and natural environmental testing we are afforded an opportunity to continually study and understand the mechanics of failure and so approach total reliability. Because of the flexibility, and relative low cost of simulated testing as compared to field testing, it is evident that the Climatic Laboratory is one of the most outstanding examples of money saving and methods of improvement within the Department of Defense.

THE BO 105 ICE PROTECTION SYSTEM

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Hans Melcher, LTC — BWB, Federal Office for Military Technology and Procurement of the FRG

ABSTRACT

Some remarks are given on the German military requirements for helicopter ice protection systems. With respect to these requirements a development program, funded by the MOD (Ministry of Defence) of the FRG, was started on the H34 helicopter. This program is briefly mentioned.

Under contract of MOD, a rotor deicing system for the reinforced plastic rotor blades of the MBB BO 105 light twin engine helicopter was developed.

The program's major steps included:

- mini ice rig tests
- tests in a climatic chamber
- wind tunnel tests in the icing wind tunnel of Vienna
- spray rig tests at the National Research Council facilities in Ottawa
- flights under natural icing conditions

The BO 105 ice protection system, including engine protection, is described. The rotor deicing system consists of a DC control unit, developed by AEG-Telefunken and spraymat ice guards, fabricated by WMI, the German licensee of Lucas Aerospace.

The test results including temperature distribution in chordwise and spanwise direction of the blades are discussed. On the FAA curve of liquid water content vs. temperature (FAR 25 Appendix C), many test points according to the NRC (National Research Council of Canada) test procedure have been successfully per-

formed down to $-21^{\circ}C$. In an 8 week period, the helicopter has flown more than 30 hours under artificial and natural icing conditions.

INTRODUCTION

In the Military Regulations for flights under IMC it is stipulated, that under forecast light, moderate or severe icing conditions flights are only allowed, when anti-icing or deicing of the aircraft is provided.

As there are no production helicopters in the German Armed Forces with deicing systems, helicopter flights in forecast icing conditions are forbidden.

Therefore for an average of 20 to 30 days during wintertime helicopter flights are not allowed in Germany.

To change this situation, the German MOD started in 1968 the development of rotor blade deicing systems. The first deicing system, developed by the AEG-Telefunken Company, was installed on a test helicopter, the Sikorsky H34 (S58). After flights with the deicing system in dry air and in rainy weather under VFR at temperatures above $0^{\circ}C$ it was intended that the experimental system be tested in winter 72/73 in icing conditions on the Wasserkuppe, 950 m above sea level in the Rhön in the area of the 616th US Air Control and Warning Squadron.

During the tests (Fig. 1) the helicopter was fettered. All measured data was transmitted over cables and a telemetry system.
Although all the metereological measurement equipment was iced severely (Fig. 2), the outside air temperature (OAT) never dropped below $-7^{\circ}C$ with high LWC.

As the ice does not begin to accrete on the rotor blades of the S 58 at approximately $-6^{\circ}C$, there were no icing conditions to test ice shedding. The only thing to do was to record the temperatures occuring along the blades at different weather conditions. During all the tests, the deicing system worked electrically correct.

In 1970 the system used on the S 58 was adapted to

the BO 105 helicopter and at the end of 1974 the adaption of the BO 105 deicing system to the Sea King helicopter MK 41 of the German Navy will be started.

THE BO 105 ICE PROTECTION SYSTEM

The BO 105 is a light twin engine 5 seat multi purpose helicopter with a maximum gross weight of 2,300 kg. The main data is summarized in Fig. 3.

The test helicopter is a production version with IFR equipment and an ice protection system. This ice protection system covers:

- engine's intake anti icing system and protection
- heated pitot tubes
- electrically heated pilot's windshield
- a rotor deicing system for the fiberglass rotor blades (main and tail rotor)
- ice detectors

The engine air intake is protected by a so-called snow and ice deflector in front of the plenum chamber (Fig. 4). The air, before entering the plenum chamber, is deflected 90°C downward in front of the main rotor gear box by this deflector. Thus snow and ice particles are led to the engine deck. The bleed air heated compressor intake behind the gear box 160 inches above the engine deck does not suck in snow or ice.

The pilot's windshield, supplied by Lucas Aerospace is electrically heated with AC current 3 phases (Fig. 5). The specific power is 2.8 watts/in^2.

The present rotor deicing system (main and tail rotor)is specified to operate down to -18°C and has a total weight of 12 kg not including power supply and wiring. The heater elements are of spraymat type, fabricated by Westfälische Metallindustrie (WMI), the German licensee of Lucas Aerospace, UK. The DC control system is developed by AEG-Telefunken, Germany. Each rotor blade has 6 spanwise heater elements in the leading edge under the erosion protection shield,

which are powered with 260 VDC.

Figure 6 shows the spanwise specific power distribution on the fiberglass rotor blades. The area, covered by the ice guards is 27% chord on the lower side and 15% chord on the upper side. This coverage is very high and there are provisions to reduce the number of heater elements energized during the test campaign to 3 in order to find the optimum coverage, giving a protection of only 13.5% on the lower side and 8.5% on the upper side. In two opposite blades of the four bladed rotor the corresponding mats are energized simultaneously to maintain a symmetric ice shedding (Fig. 7).

The tail rotor has also 6 heater elements with a specific power of 2.64 W/cm^2.
The coverage is 18% on the lower and 12% on the upper side. All mats are energized simultaneously with every 4th step of the heating cycle.

The heater ON time can be selected by hand continuously variable between 2 and 24.5 seconds and the heater OFF time between 0 and 130 seconds.

The airfoil section of the main and tail rotor blades does not differ from the production blades. The heater mats are built into a recess in the blade's surface.

Fig. 8 shows the heating sequence of the elements of main and tail rotor.

The AEG-Telefunken DC control system consists of the following components:

- The control unit, mounted in the right rear side of the fuselage between the frame behind the fairing, is shown in Fig. 9.
  Fig. 10 shows the two development stages of the control unit. Two years ago there were two components with a total weight of 32.7 kg. The current system weights 4.7 kg only and incorporates

  o the thyristor rectifiers to rectify the alternators power in 260 V DC

- o the module for controlling the heating sequence of the mats
- o the thyristor contactor switching the power ON and OFF
- o the monitoring system for over and under current, short circuit, sequence control and element ON time

in a single module (Fig. 10 lower box right hand side).

- The cockpit mounted control panel for switching ON and OFF the rotor deicing system is shown in Fig.11. It incorporates the knobs for the selection of heater ON time and OFF time. The OFF time is the time between the end of a heating cycle and the beginning of the next heating cycle.

- The electronic stepping switch on the main rotor head distributing the power to the heater elements in the blades is illustrated in Fig. 12.

- The slip ring assemblies with 3 slip rings on the main rotor and 2 slip rings on the tail rotor for the DC power transfer is illustrated in Fig. 13. In the experimental system the main rotor slip ring assembly incorporates 22 additional slip rings used to data transfer.

Fig. 13 shows also the two development stages for the tail and main rotor slip rings, the stepping switch and the control panel. The reduction of weight and volume was possible by using new electronic components. The efficiency is 93-95%.

The installation areas are shown in Fig. 14. The power supply is accomplished by two alternators, each driven by one engine.

One alternator supports the rotor deicing system. The power required including tail rotor is 5.7 KW. The other one supports the cabin heating (4,0 kVA) and the electrically heated windshield (2.1 kVA). In an emergency case, e.g. one engine inoperative, one alternator feeds the deicing system, half of the cabin heating and the lower part of the heated windshield.

The production helicopter with a deicing system will have one 10-12 kVA alternator directly driven by the main rotor gear box.

The weight breakdown of the present test version and the production version is given in Fig. 15.

In the production version the complete ice protection system including one alternator, will come to about 32.5 kg.

Four ice detectors had been installed. One Rosemount and one Teddington on the roof in front of the engines' air intake and one Teddington and one Rosemount on the lower front part of the fuselage.

Fig. 16 shows the parameters to be measured during the tests.

TEST RESULTS

The engine intake deflector was optimized in a mini ice rig at the MBB facilities (Fig. 17). The upper control system including swash plate was tested in a climatic chamber (Fig. 18).

The function of the helicopter excluding main rotor blades has proved well in the icing wind tunnel of Vienna (Fig. 19).
Although the tail rotor had no deicing equipment during these tests, no undue strain was measured.

During all test runs no power loss of the engines and no critical ice accretion on the air intake of the engines was observed.

In Feb. 1972 the test helicopter, additionally equipped with a rotor deicing system, was tested in the ice rig of the National Research Council of Canada (NRC) in Ottawa (Fig. 20).
The wind tunnel test results concerning engine intake, control and fuel system could be verified. The electrically heated windshield proved effective.

The test results led to a complete re-design of the rotor deicing system on the blades as well as on the electronic control system.

The present system was tested from December 1973 through April 1974.
All our efforts were concentrated on testing the rotor deicing system because the other problems concerning helicopter operation under icing conditions had been solved in former test campaigns.

Objectives

The basic objective of the 1974 Canada Test Program was to demonstrate compliance with the FAA and CAA requirements.
The FAA (FAR 25 Appendix C) requirements are:

- 17.4 nm for maximum continuous
  i.e. 10.45 min. at 100 kts.
- 2.6 nm for maximum intermittent
  i.e. 1.55 min. at 100 kts.

The CAA requirements applicable for Great Britain are somewhat more severe.
The main differences are:

- droplet diameter 20 µm
- minimum flight time 1 hour or an ice accretion of 1 inch at critical parts of the helicopter.

In the spray rig the tests were made with a droplet diameter of 30 µm.
The flight time was 30 minutes, because due to the NRC experience the equilibrium between icing and deicing is attained after that time.
In the forward flight tests in natural icing conditions it was tried to have a minimum flight time of more than one hour in the cloud.

Temp. Distribution vs Radius on the Erosion Protection Shield, Dry Air Tests

During the test flights in Munich, the temperature distribution (Deicing OFF) on the blade's leading edge in dry air was recorded (Fig. 21).

The tip is heated up 10° due to aerodynamic heating. Temperature distribution was independent of altitude.

Between hover and forward flight at 60 kts. there is a temperature increase of about 4^{o}C over the blade length.
When flying between 60-100 kts. no temperature change was observed.

With a heater ON time of 4 sec at -3.5^{o}C OAT the max. temperature was nearly const. vs radius at a value of 8^{o}C due to the graded specific power distribution in the mats.

<u>Flight Hours under Icing Conditions</u>

Total test flight time	68 h
Flight time in artificial cloud	23 h 23'
Flight time in clouds under natural icing conditions (IMC, forward flight)	8 h 16'

The test results presented in the following are based on this flight test experience.

<u>Spray Rig Test Summary</u>

The spray rig test envelope is summarized in Fig. 22.

The FAA curve for 15 μm droplet diameter was simulated in the spray rig with 30 μm droplets in order to be on the conservative side.
The dots are test points with 6 mats energized, the triangular signs are test points with 3 mats energized per blade.
A circle around these marks means a 30 min test flight.

Some tests were made with a LWC above the FAA curve (FAR 25 C) in order to explore the limitations of the system being tested.

All the 30 min tests from -5.5 to -19^{o}C were completed successfully.

Above the temperature of -5.5^{o}C due to the aerodynamic heating of the blade, no rotor deicing system is necessary.
Even the flights 104 and 108 with their high LWC of 1.0 g/m^{3} were performed without any problems.

Limitations of the Present Rotor Deicing System

During the rig operation we found the two limits of the system being tested (Fig. 23):

- the temperature limit at $-19^{o}C$
- the limit for maximum LWC as shown in the figure with the rig droplets of 30 µm. This line represents the condition where the time between the end of one cycle and the commencement of the next one is zero.
  It can be seen that the LWC limit is beyond the FAA requirement for MAX CONTINUOUS.

The operation limit shown is a result of 30 min flights. In considering that FAA requires for MAX. INTERMITTENT (LWC as dotted line) only a flight time of 1.55 minutes at a speed of 100 kts, it may be expected this requirement to be met for a substantial part of the temperature range with the tested rotor deicing system. At temperatures higher than $-5.5^{o}C$ up to this day, no limitation was recognized.

The LWC can be raised up by reducing the number of mats energized per blade thus reducing the cycle time.

Heater ON and OFF time over OAT

The correlation of heater ON and OFF time as a function of OAT found in the rig tests based on NRC test procedure, is summarized in Fig. 24 for the book condition. The ordinate represents the OAT. The left abscissa indicates the heater ON time and the right abscissa the OFF time (Pause time) for a 6 mat and a 3 mat rotor deicing system.

For example for an OAT of $-12^{o}C$ there is a heater ON time of 12 sec. and an OFF time of 21 sec. for 6 energized mats and 38 sec. for 3 energized mats per blade.

The few 30 min tests which were made with 3 mats energized per blade were also successful with the same heater ON times as we had for the 6 mat system per blade.

The max. heater ON time of the tested AEG control system is 24.5 sec. Due to this limitation a tempera-

ture limit of the system is found to be -19°C. Another reason will be seen later.

Runback

In the temperature range from -6 to -8°C the occurence of runback was observed.
At this relatively high OAT the surface of the erosion prot. shield is not cooled fast enough below 0°C.
In this temperature range the rotor deicing system is very sensitive to the heater ON time.
The OFF time does not affect the runback phenomenon very much.
Fig. 25 shows a runback on the main rotor blade due to a too high heater ON time.
In this case (-3.9°C) as mentioned earlier, the rotor deicing system is not needed at all.
Fortunately in all the tests during which runback occured it was only observed up to a radius of 60%.

Temperature Distribution vs Radius on the Erosion Protection Shield

During some of the test flights the temperature on the blade leading edge (erosion protection shield) was recorded. Fig. 26 gives the temperature distribution vs radius for a heater ON time of 24 sec and for several other heater ON times.
Although the max. temperature of the flight with 24 sec heater ON time in the middle of the blade was between 13° and 35°C the ice was not shedded. Thus at the first glance it seems surprising that at the tip with a lower temperature the erosion protection shield was clean. But this is caused by the higher centrifugal force.

The other temperature curves for different heater ON times at nearly the same OAT and LWC show that at the inner part of the blade there is a noticeable temperature increase while at the tip there is no temperature difference between 14 and 24 sec heater ON time. This fact is the second reason for the temp. limit of the system as tested.

If the operational limit of the system was to be moved down to lower temperatures than -19°C (the present system was specified for -18°C) the specific

power between 35 - 75% radius should be increased to about 3 W/cm^2. The specific power on the tip section should be also increased by simular amounts.

Temperature Distribution vs Chord ($\frac{r}{R} = 0.5$)

The upper part of Fig. 27 shows the temperatures recorded under the heater mats in the blade vs time. The max. value is under the mat No. 3 in the leading edge: about 63 ^{o}C. The other two mats Nos. 2 and 1 on the upper side have about 53 and 45 ^{o}C. The mats Nos. 4,5 and 6 on the lower side have a lower temperature. The value is about 30 ^{o}C.
We have not made calculations for this phenomenon yet.

On the lower part of this figure there is shown the surface temperature on the erosion protection shield. The temperature on mats 1 and 3 is about 22 or 25 ^{o}C. The temp. sensor on mat 5 was unfortunately inoperative due to a malfunction. Based on other test results we know that the max. temperature on the erosion protection shield of mat 5 is about 15 ^{o}C.

A 30 Minute Flight in the Spray Rig

During the flight No. 73 in the spray rig the cyclic flapping moment $\Delta M_{\beta o}$, the cyclic control rod forces ΔSt_1, the main rotor unbalance, the rotor power required and the heating cycles were recorded vs flight time.
Flight conditions are shown in Fig. 28.

You can see that after entering the cloud the torque increases. At a 10% torque increase the deicing system was switched on and over the complete flight time the torque was stabilized within an acceptable limit. The unbalance was at a max. value of 600 grams. The alternating flap bending moment and the control rod load did not change significantly. The max. allowables are:
$\pm$ 145 mkp for flap bending moment
$\pm$ 180 kp for control loads.
Max. measured values are:
$\pm$ 35 mkp for the flap bending moment and
$\pm$ 20 kg for the control loads.
Leaving the cloud after 36 minutes the original values were obtained again.

Water Droplets

On the right side of Fig. 29 there is a probe from the rig flight 104 with an average droplet diameter of 44.7 µm.

On the left side of this picture there are shown the measured droplets during flight 164 under natural icing conditions. The diameter was 16.5 µm at $-10^{\circ}C$.

A 75 Minute Flight in Natural Icing

During flight No. 164 in natural icing LWC and droplet size, unbalance and rotor power were recorded (Fig. 30). The flight time was more than 60 minutes in the cloud at an OAT of $-10^{\circ}C$. Heater ON time was 7 sec. It is shown that an increase in LWC resulted in a torque increase. After switching on the deicing system (see bottom of Fig. 30) torque was reduced. After some heating cycles the deicing system could be shut OFF due to the drop in LWC. With a further LWC increase some cycles were started again. The unbalance force was below 250 gr. at all times. The check flight outside the cloud after 79 minutes showed nearly the same power setting as measured at the beginning on this flight.

BO 105 S2 in Natural Icing

At an OAT of $-7^{\circ}C$ at variable LWC, a 66 min test flight was performed in natural icing.
Fig. 31 shows the ice accreted on the helicopter.
The electrically heated windshield remained clean.
The following two figures demonstrate the ice accreted at the swash plate, the air intake (Fig. 32) at the empennage and at the landing gear (Fig. 33) after flight No. 135.

Flight No. 159 was performed at an OAT $-3^{\circ}C$ with the deicing system OFF. Flight time was 2 hours. Ice accretion is shown in Fig. 34 on the swash plate and the control links.

During all flights no problems were reported by the crew.
The test flights were performed by Siegfried Hoffmann, pilot, and Dieter Bender, flight test engineer.

SUMMARY OF TEST RESULTS

- The complete ice protection system worked very reliable. This statement is not only based on our flight tests. It is also a result of numerous environmental tests with the electronic system and component tests with blades equipped with ice guards. As one result of these tests a lifetime over 5000 hours for the blade with ice guards could be demonstrated. At the moment additional tests are under progress.

- During the flight tests no problems with the complete helicopter were observed:

  o main and tail rotor
  o engines
  o electrically heated windshield
  o control and fuel system
  o airframe and empennage

- No problems - as far as handling qualities and vibrations are concerned - have been reported.

- A 6% torque increase caused by ice accretion resulted in a 4% RPM reduction during autorotaion.

- No damage by shedded ice was observed.

- The electrical parameters of the ice guards (insulation, electrical resistance) did not change.

- A production version of an ice protection system will incorporate the following detector equipment:

  o OAT
  o ICE RATE
  o HOT ROD for pilot's information

- Total weight of production ice protection version is 32.5 kg.

- No rotor deicing is necessary down to $-5^{\circ}C$.

- Heater ON time is a function of OAT only.

- Pause time is a function of LWC and in second

order of OAT.

- The measured correlation for heater ON and OFF time as a function of OAT and LWC in the spray rig was duplicated under natural icing conditions.

- Runback in the temperature range -6 to -8°C at 20% to 60% radius was observed. The runback is very sensitive to heater ON time and can only be controlled by careful selection of this parameter.

- The temperature limit of the tested rotor deicing system was at -19°C.

- We recommend four energized mats per blade. The coverage should be 8% chord on the upper side of the blade and 13% chord on the lower side.

- The specific power distribution vs radius should be improved.
  A small amount of grading vs radius is of advantage as far as energy required is concerned.

- There were no problems with the tail rotor deicing. However, it can be concluded from the test results that a tail rotor deicing is necessary. A specific power of 1.6 W/cm^2 = const. vs radius seems to be sufficient.

CONCLUDING REMARKS

The BO 105 deicing system proved effective in the specified environmental conditions.

The system functions satisfactorily.
MBB will continue the development work by performing more tests under natural icing conditions and will simplify the control panel.
It is planned to certify the BO 105 for flights under icing conditions in 2 steps:

- down to -5°C without rotor deicing
- below -5°C with a rotor deicing system

All further tests will be made in Europe under natural icing conditions.

References

(1) Dipl.-Ing. Dieter Bender

Vereisungs- und Enteisungsversuche am Hubschrauber BO 105 S2 im Winter 1973/74
MBB-Bericht TN D143-156/74 vom 24.4.74

(2) Specification of the Lucas/WMI ice guards for the BO 105

(3) Spezifikation der AEG-Telefunken Rotorenteisungs-steueranlage für die BO 105

Fig. 1 Fettered S58 on the Wasserkuppe

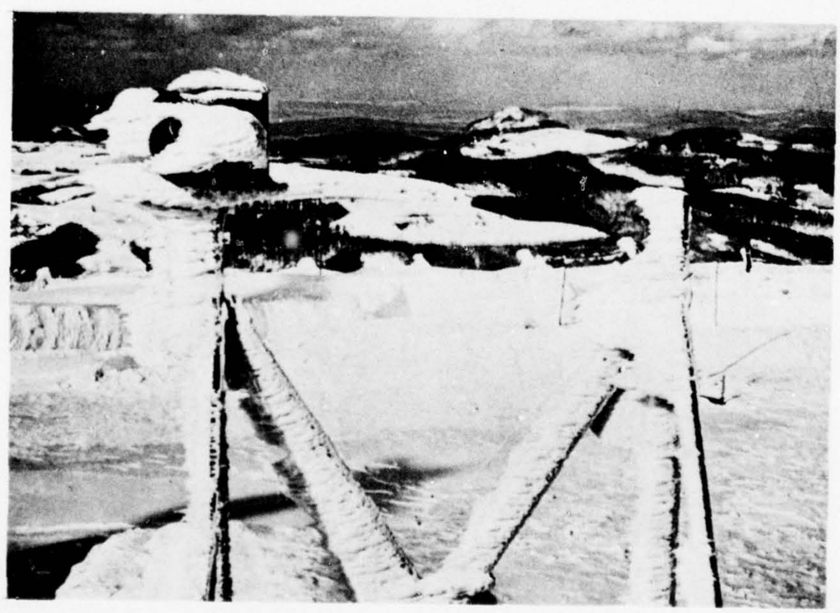

Fig. 2 Severely Iced Metereological Equipment

AD-A061 423 ARMY AVIATION ENGINEERING FLIGHT ACTIVITY EDWARDS AF--ETC F/G 1/3
ROTARY WING ICING SYMPOSIUM. SUMMARY REPORT. VOLUME III.(U)
JUN 74 D E WRIGHT
UNCLASSIFIED USAAEFA-74-77-VOL-3 NL

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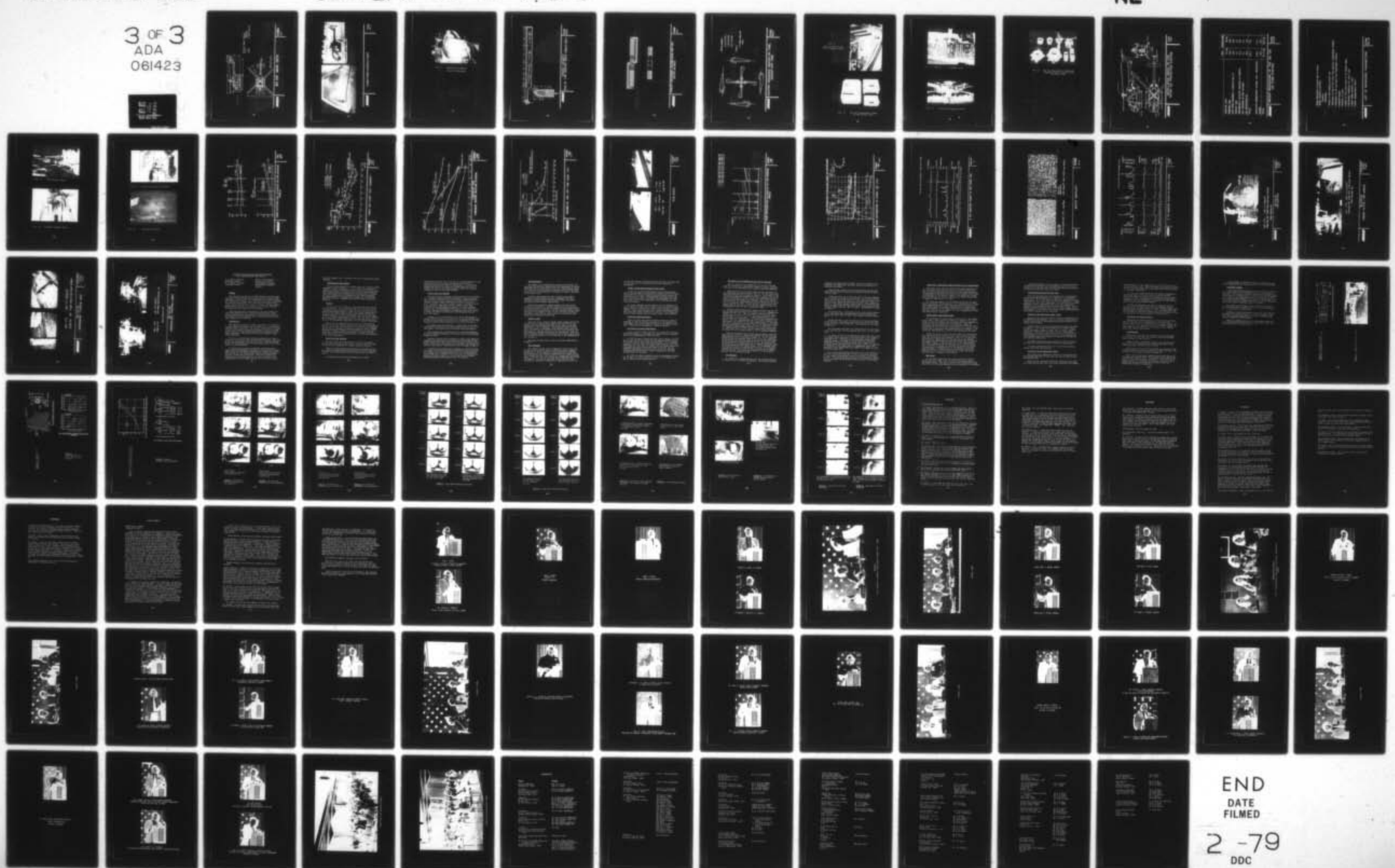

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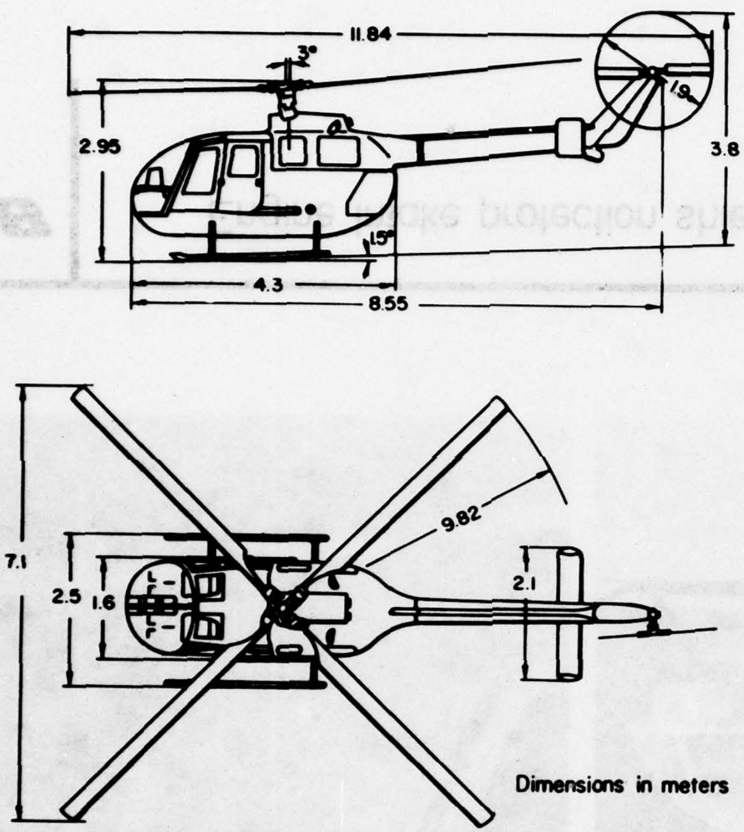


Max Cross Weight =2300 kg
Cruise = 250 km/h
155 m /h
Engines =2 allison 250 C 20

MBB

BO 105 MAIN DATA

ICING

FIG. 3

MBB

Engine intake protection shield

Icing
FIG. 4

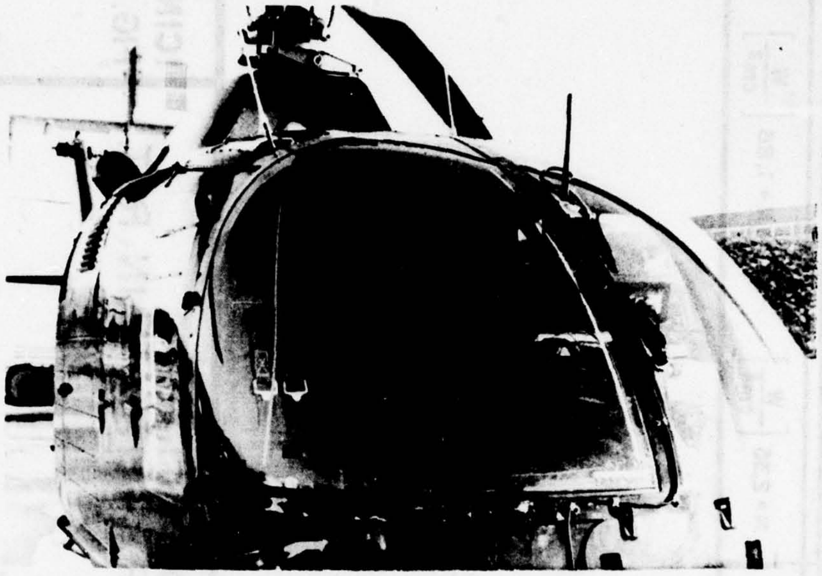
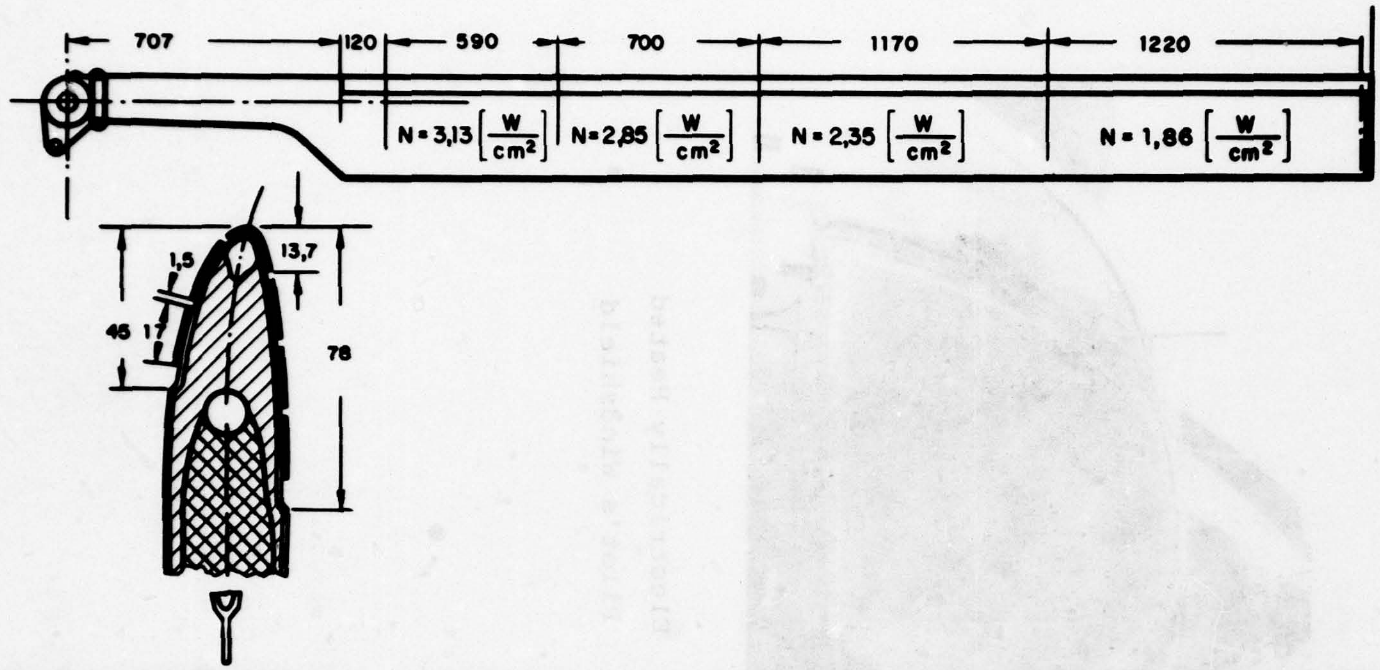


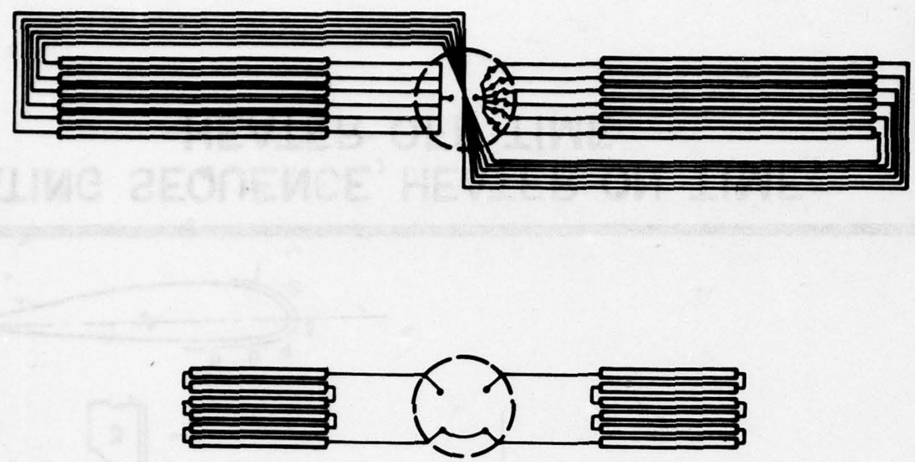
Fig. 5 Electrically Heated Pilot's Windshield



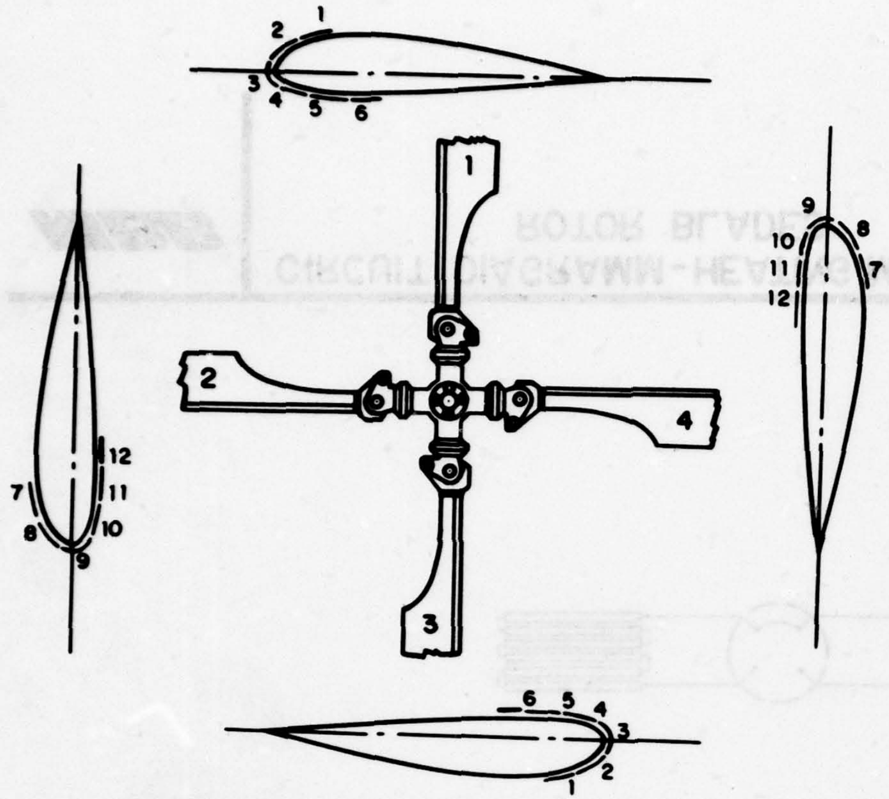
MBB

POWER DENSITY
POS. OF THE HEATER MATS (MAIN ROT.)

ICING
FIG. 6



MBB | CIRCUIT DIAGRAMM - HEATING MAIN AND TAIL ROTOR BLADES | ICING FIG. 7



Heating sequence :

3, 4, 2, 5+ Tail,

1, 6, 9, 10+Tail,

8, 11, 7, 12+Tail

1 Cycle =

12 Heater steps and

Pause

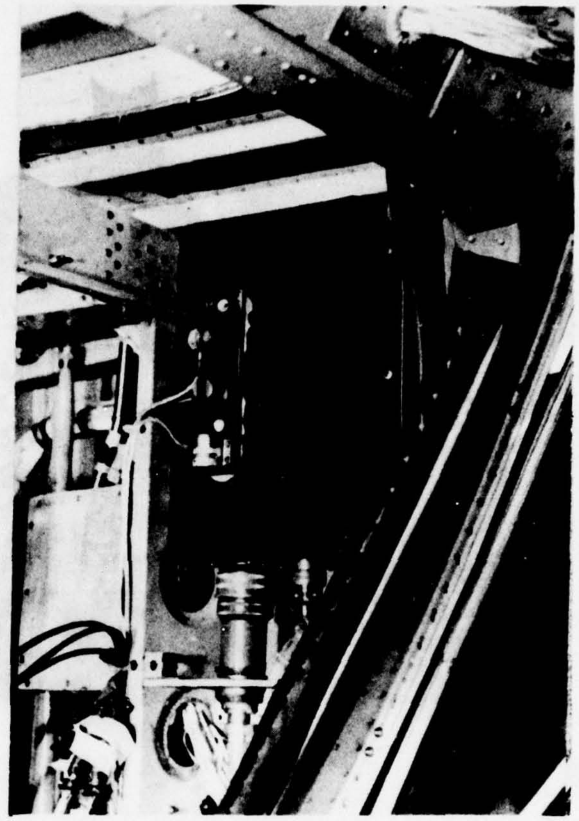
MBB

HEATING SEQUENCE, HEATER ON TIME,
HEATER OFF TIME

ICING
FIG. 8

Fig. 9

Control Unit of the
Rotor Deicing System



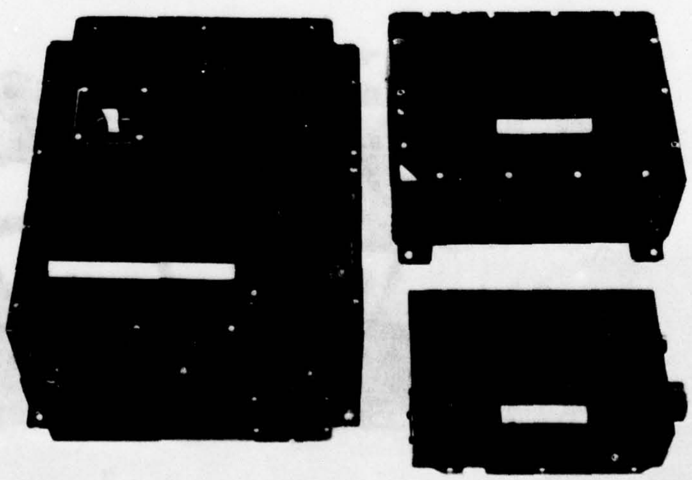


Fig. 10 The Two Development Stages
of the Control Unit

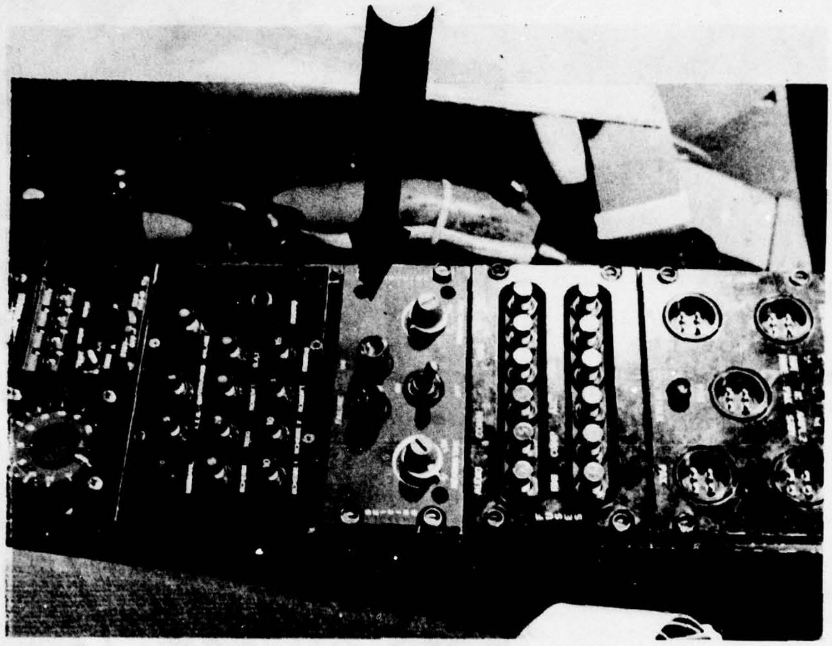


Fig. 11 Control Panel

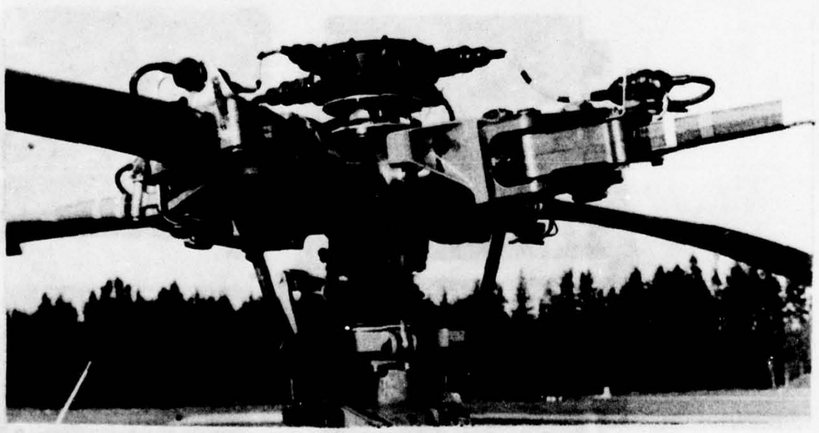


Fig. 12 Electronic Stepping Switch

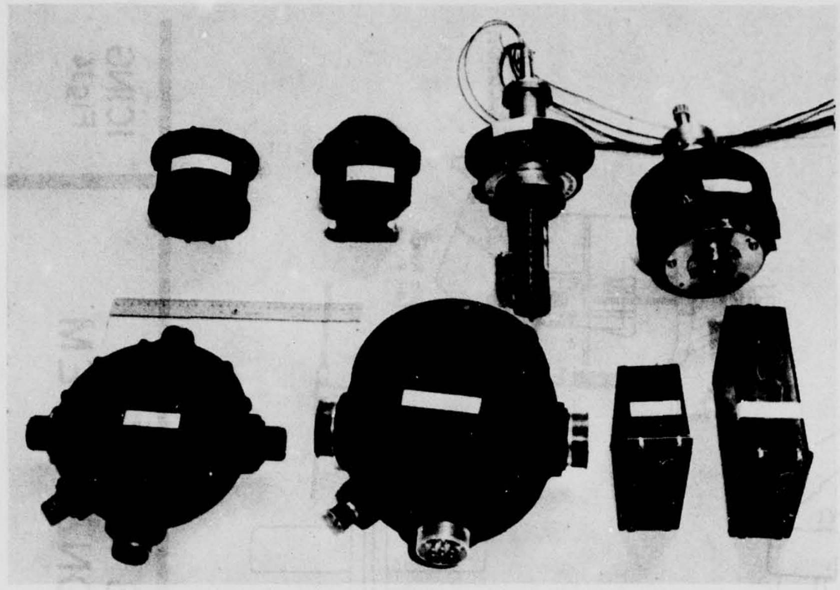
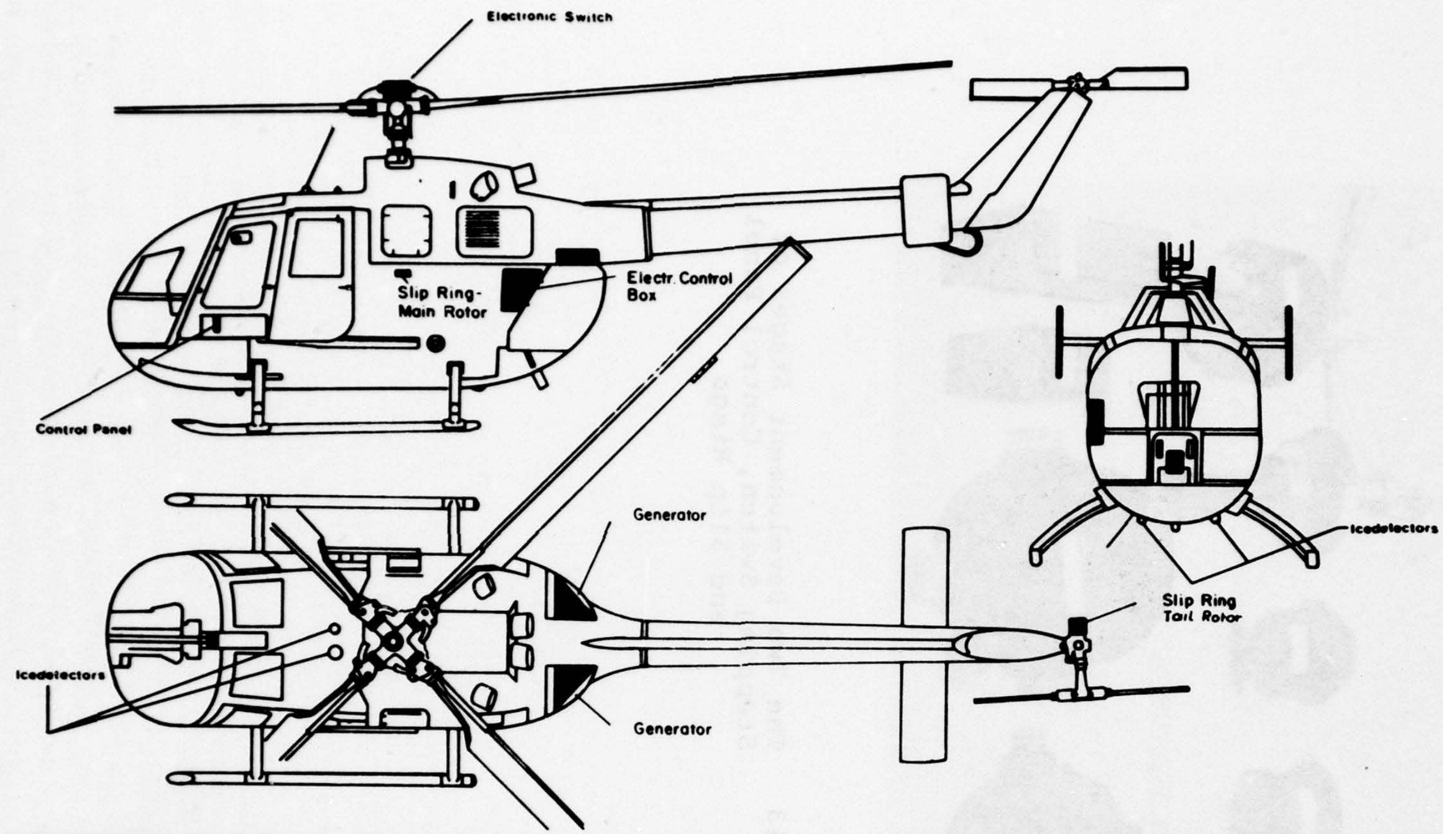


Fig. 13 The Two Development Stages of Stepping Switch, Control Panel and Slip Rings



MBB

INSTALLATION AREAS OF THE
ROTOR ICE PROTECTION SYSTEM

ICING
Fig.14

	TEST	PRODUCTION
CONTROL UNIT	4,1 kg	4,1 kg
CONTROL BOX	0,6	0,6
SLIP RING ASSEMBLY (MAIN & TAIL R.)	4,2	4,2
SWITCH BOX	3,1	3,1
WINDSHIELD (ELECTR. HEATED) △ WEIGHT	6,2	3,5
ICE - GUARDS IN THE BLADES	0,1	0,1
ICE DETECTOR (S)	5,0	1,8
TOTAL	23,3	17,4
AUXILIARY GENERATOR SYSTEM INCL. MOUNTINGS	25,0	11,6
WIRING	6,0	3,5
TOTAL	54,3 kg	32,5 kg

MBB

WEIGHT BREAKDOWN OF THE BO 105
DE - ICING SYSTEM

ICING
FIG.15

- Temp. distribution vs radius and vs chord
- Blade bending moments
  - · flapwise
  - · chordwise
- Forces in the rotating control rod
- Vibrations on top of the main rotor gearbox (longitudinal, lateral)
- Vibrations on tail rotor gear box (longitudinal, lateral)
- Engines' torque
- Position of pitch
- Heater on time and amps of the heater steps
- Centrifugal unbalance on main rotor
- Liquid water content (LWC)
- Outer temperature (OAT)

MBB | LIST OF MEASURED PARAMETERS | ICING FIG. 16

Fig. 17 Mini Ice Rig Tests

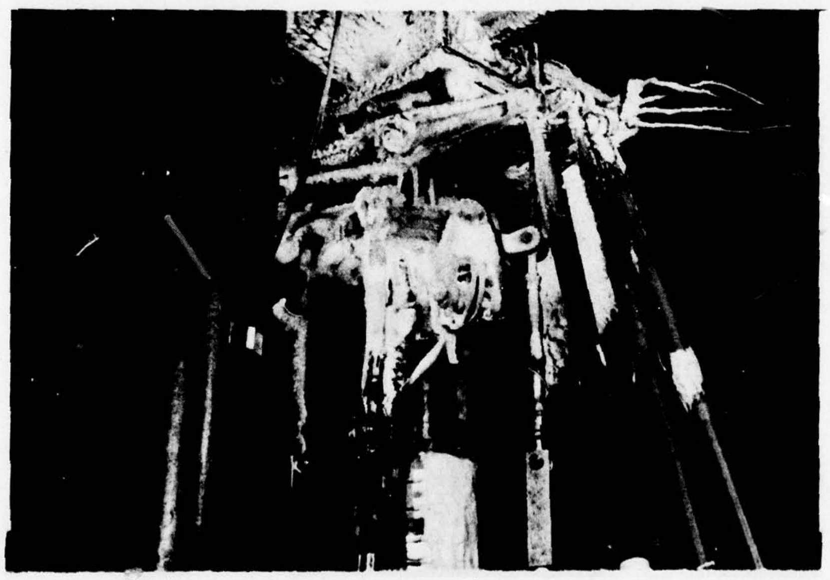


Fig. 18 Climatic Chamber Tests

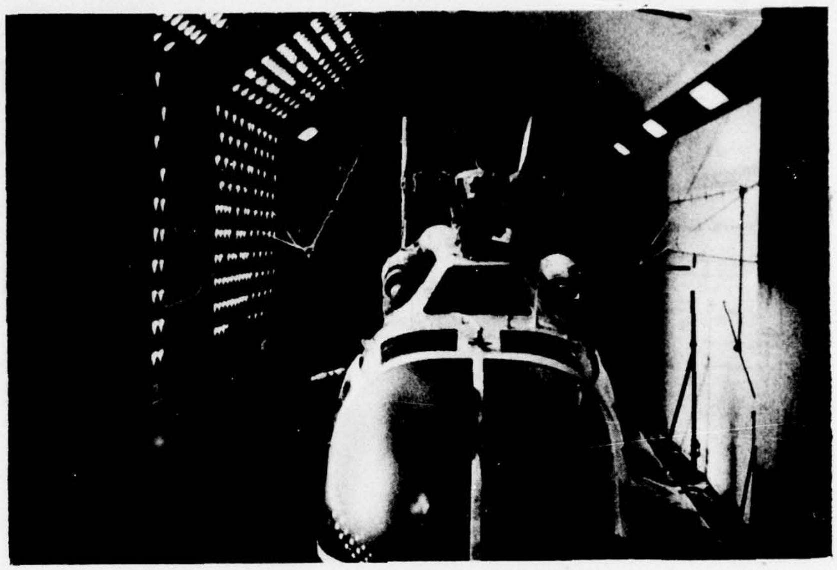


Fig. 19   Icing Wind Tunnel of Vienna

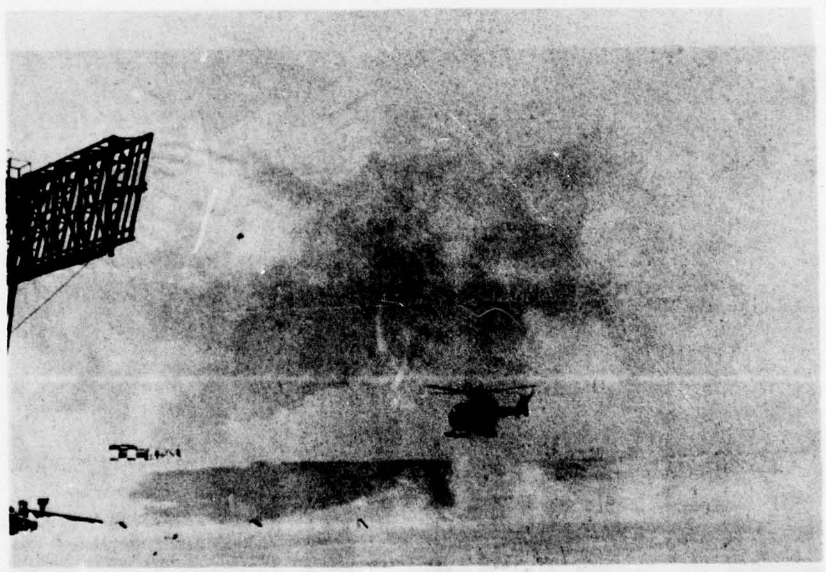
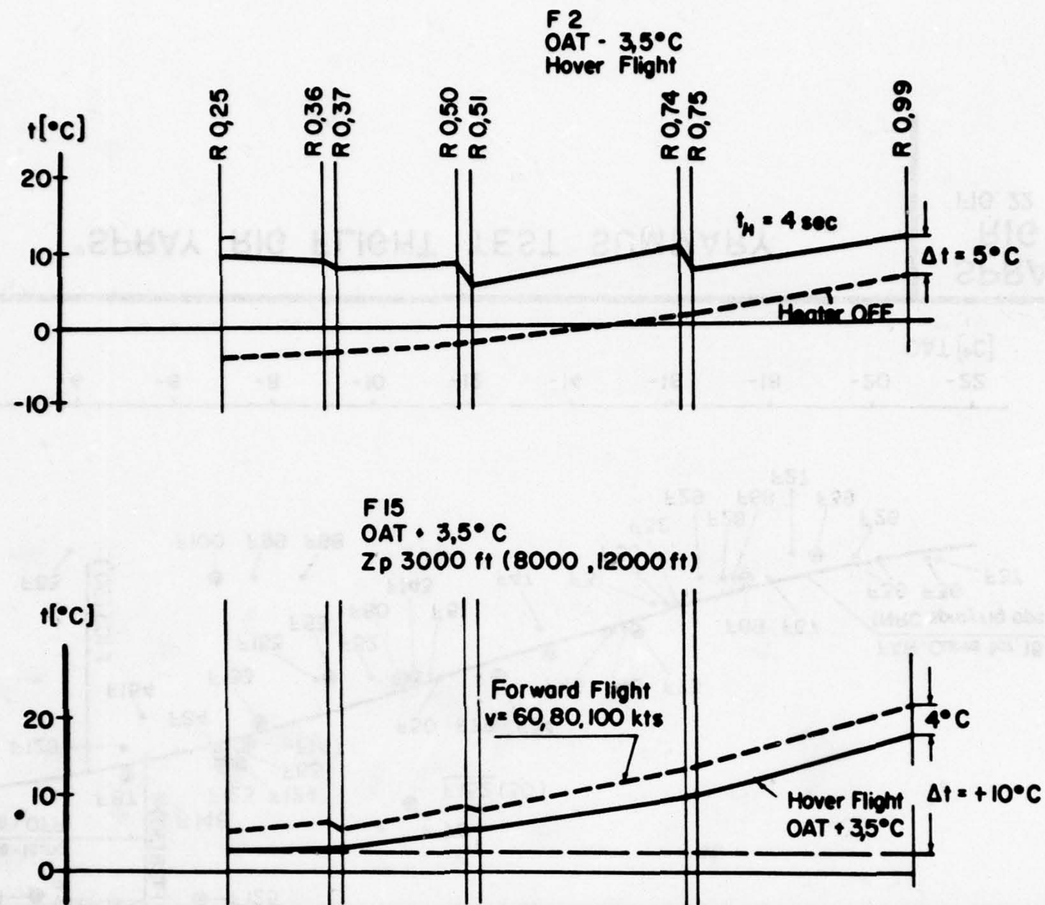


Fig. 20   Ice Rig in Ottawa

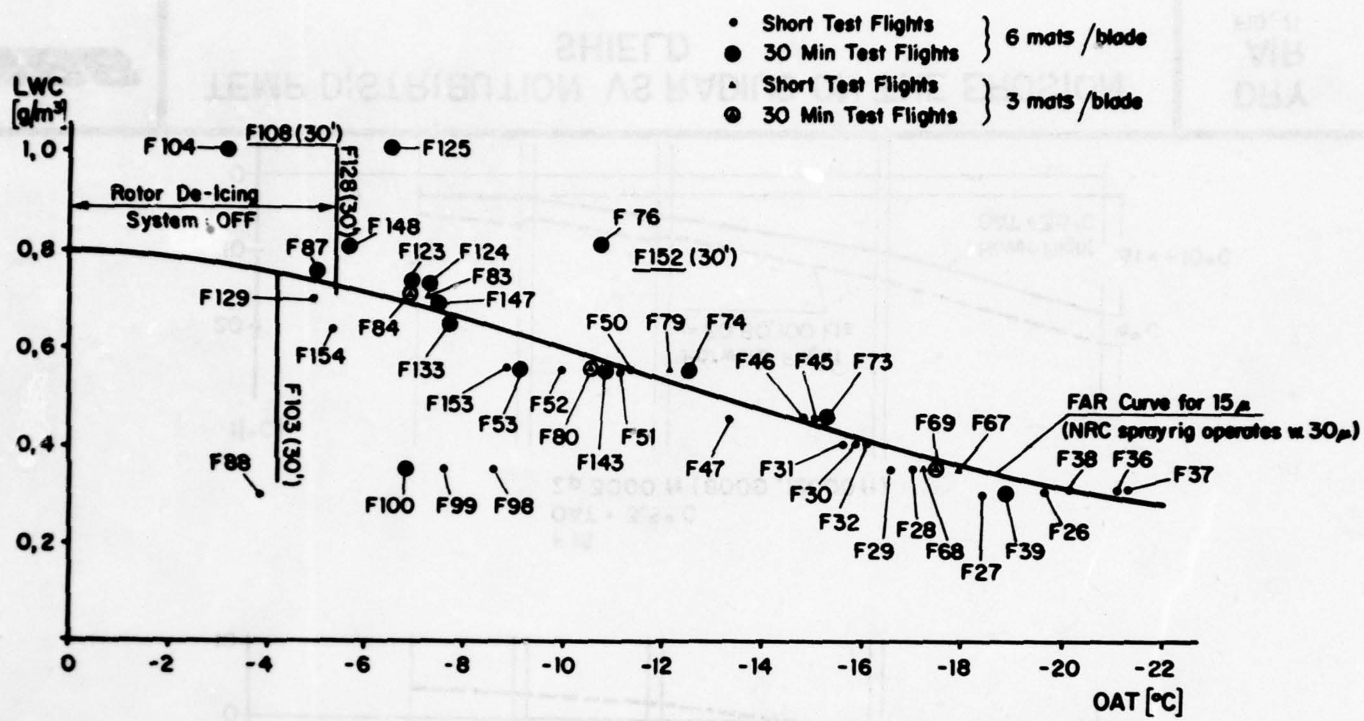


MBB

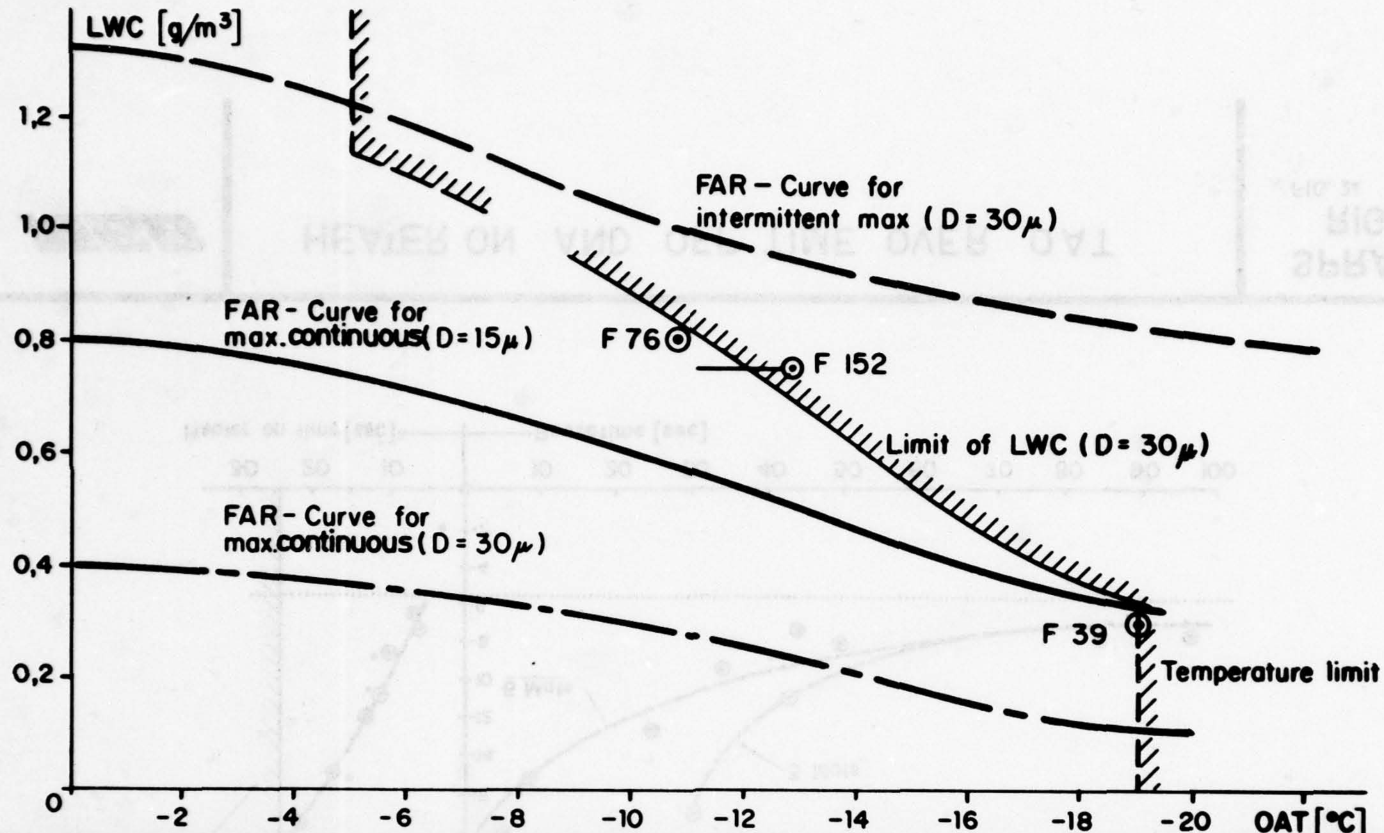
TEMP. DISTRIBUTION VS RADIUS ON THE EROSION SHIELD

DRY AIR

FIG. 21



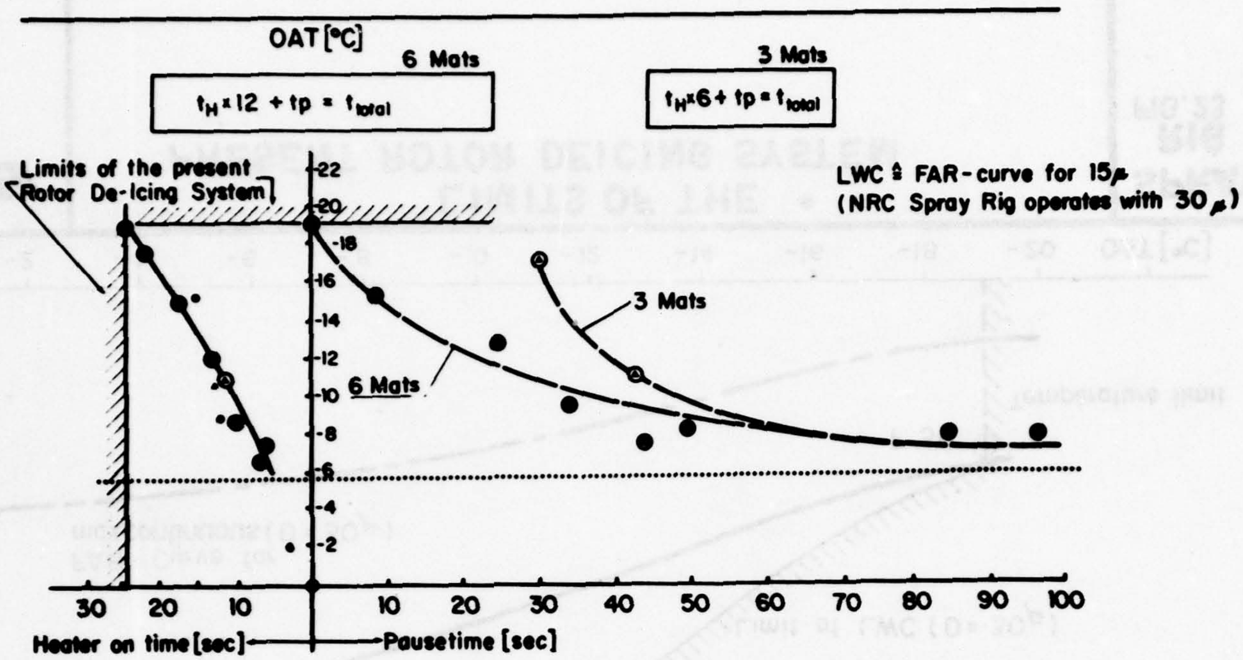
MBB	SPRAY RIG FLIGHT TEST SUMMARY	SPRAY RIG FIG. 22



MBB

LIMITS OF THE
PRESENT ROTOR DEICING SYSTEM

SPRAY RIG
FIG. 23

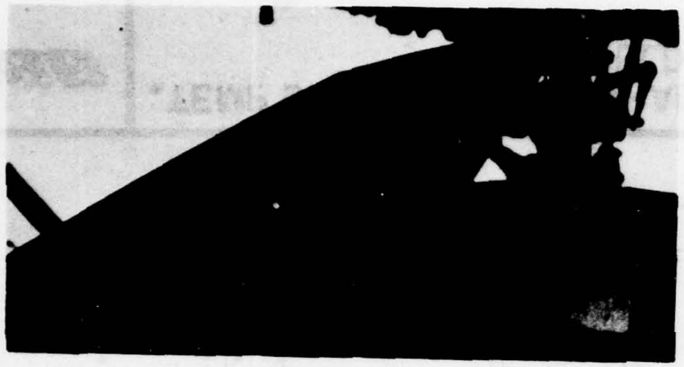


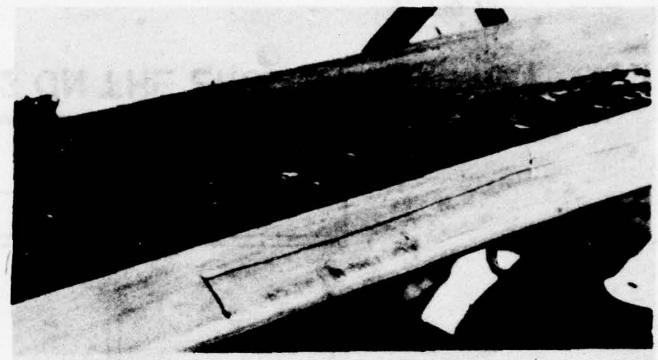
MBB

HEATER ON AND OFF TIME OVER OAT

SPRAY RIG

FIG. 24





OAT = $-3.9°C$ t_H = 4 sec

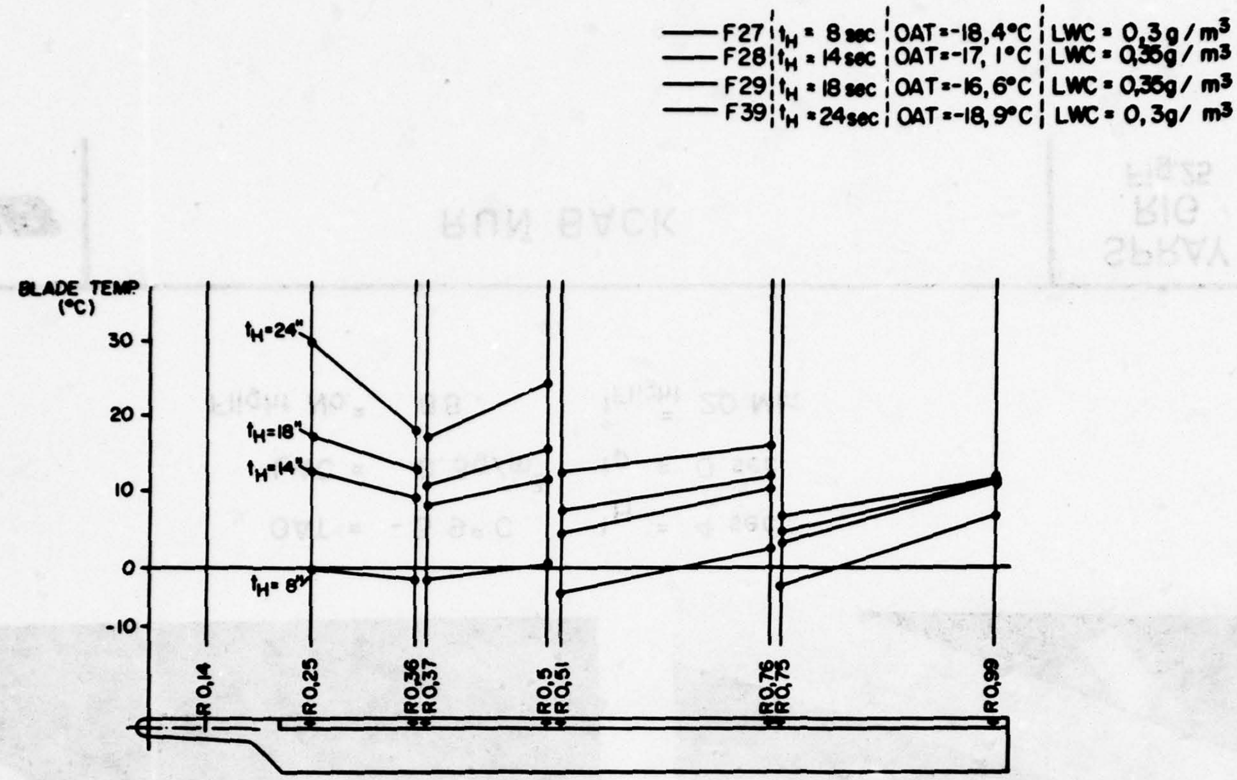
LWC = $0.3 g/m^3$ t_p = 0 sec

Flight No.= 88 t_{Flight} = 20 Min

MBB

RUN BACK

SPRAY RIG
Fig.25

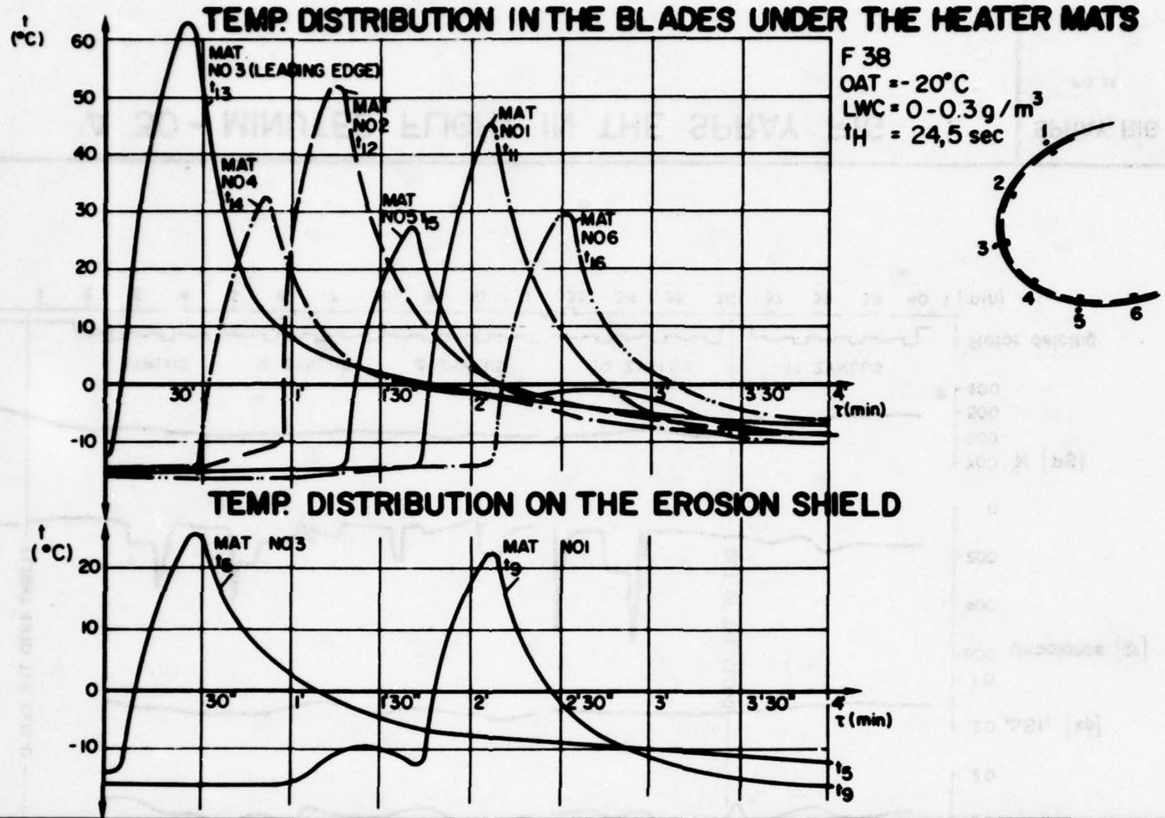


MBB

TEMP. DISTRIBUTION VS RADIUS ON THE EROSION SHIELD

SPRAY RIG

FIG. 26



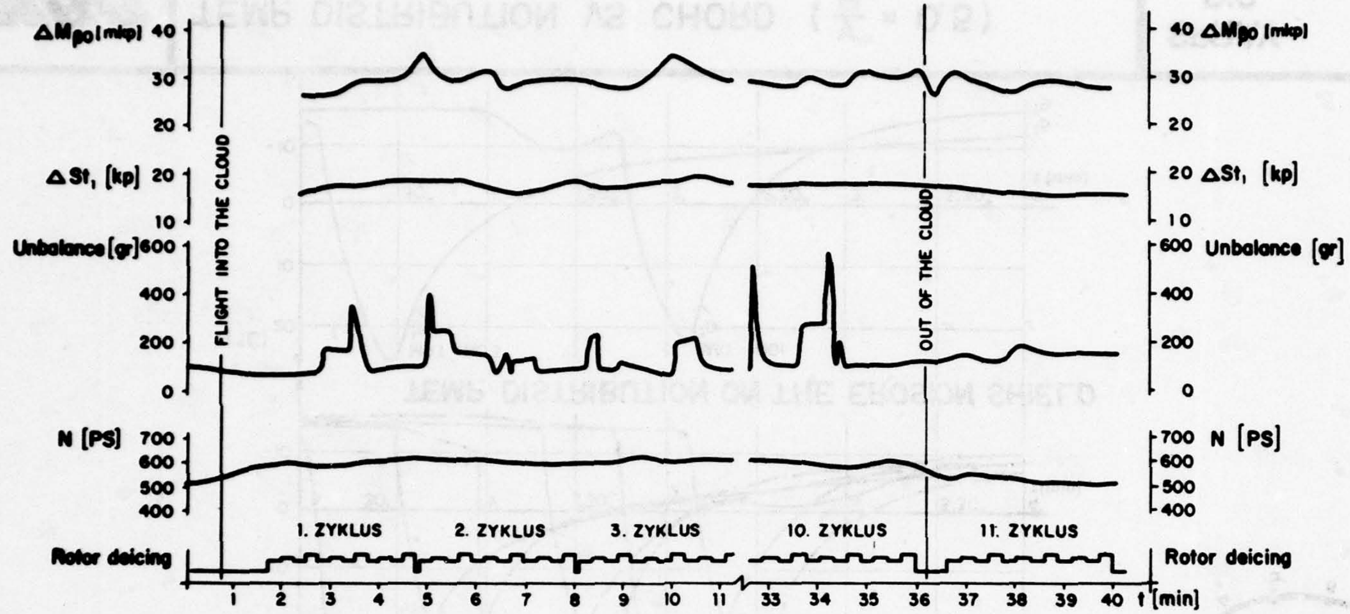
MBB

TEMP. DISTRIBUTION VS CHORD ($\frac{Y}{R}$ = 0.5)

SPRAY RIG

FIG. 27

F = 73; OAT = -15,1°C; LWC = 0,45 g/m³; t_H = 15,5-17sec; tp = 9sec

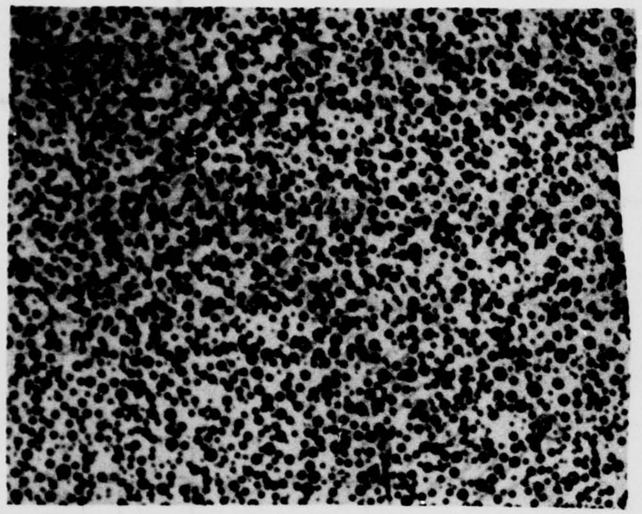


MBB

A 30 - MINUTES FLIGHT IN THE SPRAY RIG

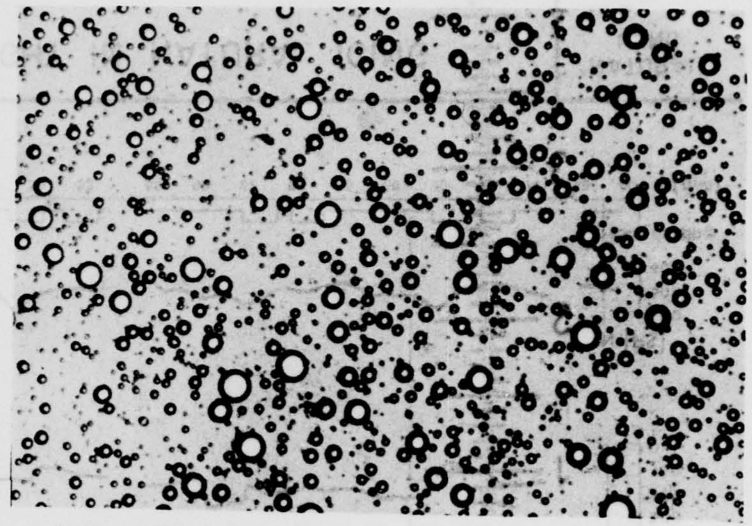
SPRAY RIG

FIG. 28



Natural Icing
Flight No 164

LWC = 0 ÷ 1,3 g/m^3; OAT = - 10°C ; D_m = 16,5 μ



Spray Rig
Flight No 104

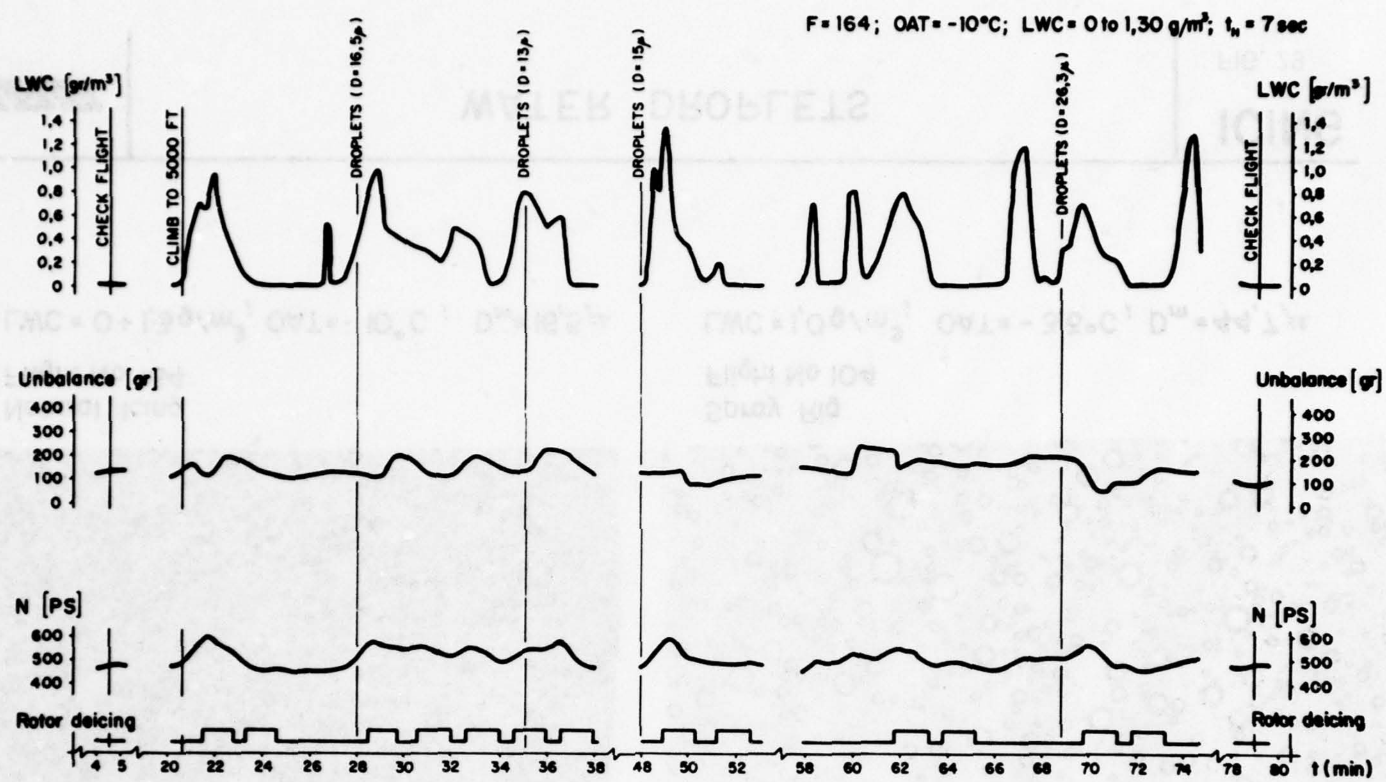
LWC = 1,0 g/m^3; OAT = - 3,3°C ; D_m = 44,7 μ

MBB

WATER DROPLETS

ICING

FIG. 29



MBB

A 75-MINUTES FLIGHT IN NATURAL ICING

NATURAL ICING

FIG. 30

OAT = -7° C   LWC = 0 ÷ 0,6 g/m³

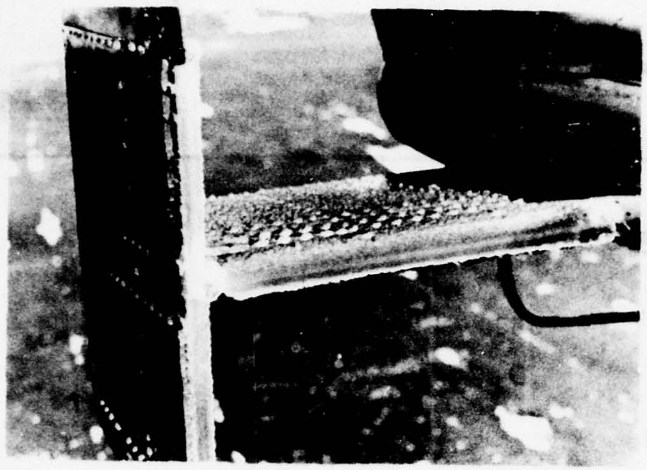
Flight No.: 135   Flight Time in the cloud 66 Min

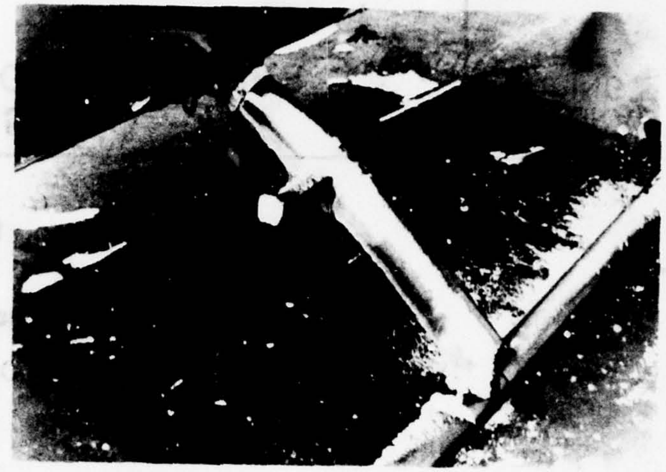
Rotor Deicing = ON

MBB	BO 105 S2 HELICOPTER	NATURAL ICING Fig. 31

OAT = -7°C, LWC = 0-0.6 g/m^3
Flight No.=135, Flight Time in the Cloud = 66 Min
Rotor Deicing = ON

MBB	BO 105 S2 SWASH PLATE, AIR INTAKE	NATURAL ICING Fig.32

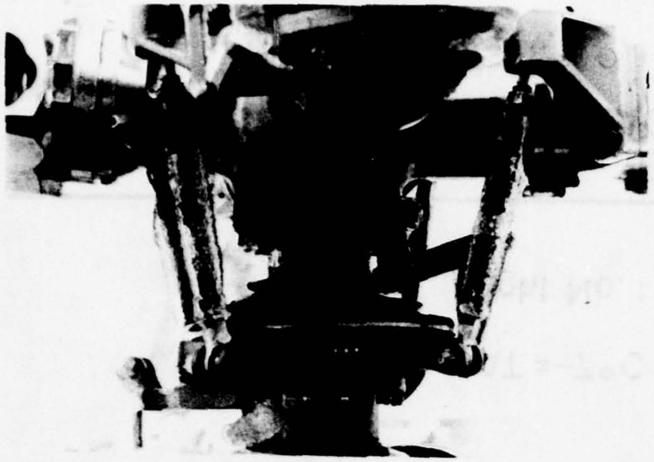


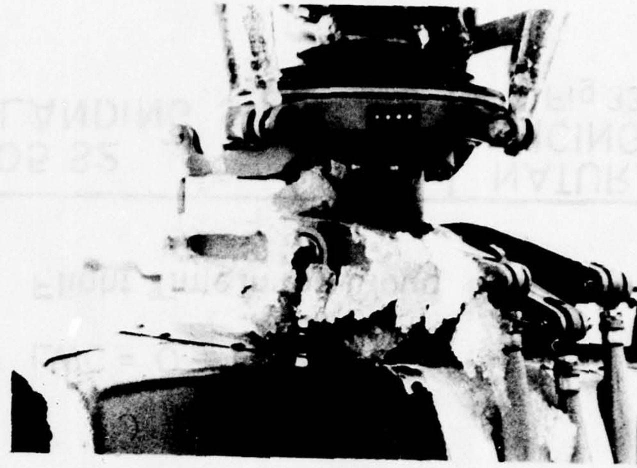


OAT = -7°C   LWC = $0 \div 0{,}6\ g/m^3$

Flight No. : 135   Flight Time in the Cloud 66 Min

MBB	BO 105 S2 EMPENNAGE LANDING SKID	NATURAL ICING Fig.33





OAT = - 3° C   LWC = 0 ÷ 1,0 g/m³

Flight No. 159   Total Flight Time in the cloud : 2h

Rotor Deicing = OFF

MBB	BO 105 S2 SWASHPLATE, CONTROL LINKS	NATURAL ICING Fig.34

SIMULATED ICING TRIALS ON HELICOPTER FUSELAGES IN AN ALTITUDE ENGINE TEST FACILITY

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Summary

The altitude test facility at NGTE includes a cell for testing large turbofan engines over a wide range of simulated flight conditions. In this cell, 25 ft diameter, front sections of full-scale helicopter fuselages less rotor blades, may also be installed for free-jet icing trials. These can be made at forward velocities greater than 50 ft per second at minimum altitudes varying between 2,500 and 8,000 ft, depending on the forward velocity. No attempt is made to simulate rotor downwash.

This Paper presents a description of the cell, defines its icing test capabilities and records observations made during trials on Sea King and Lynx fuselages to determine the effect of air temperature, water concentration, aircraft attitude and forward velocity on ice accretion.

Introduction

The Engine Test Facility at NGTE consists of five test cells in which aircraft powerplants can be tested over a wide range of simulated flight conditions. The largest and most recent of these is Cell 3 West which was originally designed and built for connected testing of large turbofan engines under simulated subsonic flight conditions. Tests may also be made in simulated icing cloud conditions.

This cell is sufficiently large to accommodate the front portion of a Sea King or Lynx size helicopter fuselage, less rotor blades, and by mounting it at the outlet from a free-jet blowing nozzle, the windscreen, cabin roof and engine intakes can be subjected to icing conditions.

To confirm that this test arrangement was satisfactory, static pressure measurements were made on a one-twentieth scale model of the cell helicopter installation to compare the pressure distribution over the fuselage upper surface with measurements made in a conventional wind tunnel. Very good agreement was obtained and it was concluded that the test cell gave a close representation of free flight, and such full scale data as are

available support this. No attempt was made to simulate the rotor downwash.

Description of cell layout

The cell configuration with a Lynx helicopter fuselage section installed is shown in Figure 1. Air is drawn from atmosphere through a cooler, inlet pressure control valve and ductwork, water spray rake assembly and blowing nozzle into the working section. After flowing over the helicopter the air passes into an exhaust duct and is pumped back to atmosphere by the plant exhausters. An inbleed valve in the exhaust duct enables the cell altitude pressure to be set. A general view of Cell 3 West is shown in Figure 2.

Cooler

The cooler is 61 ft long with cross sectional dimensions of 27 ft by 29.5 ft. It is a gas-over-tube type consisting of 33 horizontal modules divided into three sections each with its own coolant flow control system to enable the coolant temperature to be set at the entry to each section. The coolant, 33 per cent aqueous ammonia, is held in a pre-cooled store of 500 tons capacity, at a minimum temperature of -50°C. The cold store is cooled by a conventional vapour compression refrigeration system with 1 HP and 2 LP compressors using Freon 22 as the primary refrigerant.

The cooler was designed to give an air off temperature of -37°C at an air mass flow of 800 lb/s with inlet air conditions of 7°C and 100 per cent relative humidity. Approximately 30 minutes testing time is achieved at design conditions but longer running times are achieved at higher air off temperatures or lower airflow rates. Moisture in the incoming air is deposited on the cooler tubes and as this builds up there is a small deterioration in the cooler performance.

Inlet valve and ductwork

The inlet valve is a louvre type in a duct 12 ft square located just downstream of the cooler. It has 12 pairs of vanes on six shafts all connected to a single hydraulic actuator.

Three wire mesh screens are positioned at intervals in the cell approach ductwork between the inlet valve and the spray rake to remove any airflow maldistribution introduced by the louvre valve.

To simulate free cloud icing conditions it is first

necessary to condition the air so that it is fully saturated at the static condition as it leaves the blowing nozzle prior to injecting water droplets from the spray rakes. In some circumstances this entails the addition of water vapour and this is effected by means of the steam re-humidification grid situated between the first two wire mesh screens.

Water spray rake assembly

The final step in achieving icing simulation is the injection of a controlled quantity of water droplets of a given size, and this is done by the water spray rake assembly shown in Figure 3. Seventeen equally spaced stainless steel struts of $1\frac{3}{4}$ in. $\times$ 11/16 in. aerofoil section are mounted across a circular duct of 93 in. diameter. Each strut carries a number of air blast atomisers, facing downstream and spaced 5 in. apart, giving a total of 242. Interchangeable water jets of 0.014, 0.016 and 0.024 in. diameter are available and blanking caps can be fitted to reduce the number of nozzles for tests requiring very low water flow rates.

Because of the large diameter of the spray rake assembly separate water supplies are provided for the top and bottom halves so that the water flow can be balanced to compensate for the head difference which becomes significant at low water flow rates.

The spray rakes are installed at a small angle to the horizontal plane to ensure that any water remaining at the end of a test drains to the lowest point to minimise freezing problems.

The atomising air is pre-heated up to a maximum temperature of 200°C to prevent freezing at the nozzles due to the isentropic temperature drop.

The spray nozzles were calibrated in a small icing tunnel at atmospheric air pressure with airspeeds of 200 ft/s and 300 ft/s. Droplet samples were taken using the oiled slide technique over a range of water flows and atomising air pressures, and results for the 0.016 and 0.024 in. diameter water jets are shown in Figure 4.

Tests have also been made more recently in a small altitude test tunnel, where the water spray characteristics of the nozzles were measured at airstream pressures down to 5 lb/in² absolute. It was found that, at similar water flow rates and atomising air pressures the mean droplet size of the spray was not significantly influenced by the ambient static air pressure.

Instrumentation

The steady state instrumentation system has facilities for measuring 300 individual pressures and 200 temperatures as well as quantities such as shaft speeds, fuel flow, displacements, areas, etc. The data are acquired and processed on-line by an SDS 9300 computer with selected parameters displayed on digital indicators in the control room.

Transient measurements may also be made of main engine parameters, de-icing heater mat power consumption etc and recorded on either magnetic tape or paper charts, the latter providing for direct visual observations.

Four closed circuit TV cameras having a scan rate of 25 frames/s are available for general observation of ice build-up, but this scan rate is too slow to monitor the trajectories of ice as it sheds into the airstream. For this purpose a special camera scanning at 100 frames/s is used to view the region in front of the engine intakes and the output from this plus any one other camera can be recorded on video tape.

Range of tests

Tests may cover a range of flight speeds between 50 ft/s and 270 ft/s with air temperatures between +5°C and -37°C and the largest free-jet blowing nozzle that can be used to meet these conditions is 100 in. diameter. The minimum altitude at which tests can be made is determined by the pressure drop across the cooler and the wire mesh screens in the cell inlet ductwork, and this depends on forward velocity and air temperature. With the 100 in. diameter blowing nozzle the altitude varies between 2,500 ft at 50 ft/s and 8,000 ft at 270 ft/s with an air temperature of 0°C, (Figure 5).

The upper altitude limit is set by the plant exhausters at 50,000 ft.

Test technique

The forward velocity and air temperature for the test conditions are set with the intake and engine de-icing equipment switched on or off as required. Icing is started by injecting water droplets at the required concentration for a period of up to 30 minutes duration. The test cell is brought back to ambient static conditions, the cell entered, and photographs taken of the ice deposits. A return is then made to the test conditions followed by a gradually increasing air temperature to allow the

ice melt-off sequence to be observed and recorded. The time taken for the melt-off is a function of the outside ambient air temperature.

Details of Sea King helicopter icing trials

As part of a general investigation into helicopter power plant icing problems, a Sea King front fuselage less rotor blades and fitted with an early standard of mushroom type intakes, was installed in Cell 3 West. The parameters investigated were the ones thought most likely to influence ice deposition, and these were: time in icing, air temperature, water concentration, and aircraft attitude.

To obviate possible engine damage due to ice ingestion the initial tests were made with air driven ejectors fitted into the engine nacelles to provide the correct airflow through the engine intakes. Later tests were made with a Gnome H1400 engine installed in the port nacelle instead of an ejector. This allowed the effect of heat transfer from the engine carcase to be observed.

Tests with mushroom intakes

Tests were made with mushroom intakes having two different standards of anti-ice heating, referred to as "A standard" and "B standard", the latter having no heating on the mushroom front face. The heater mat arrangement is shown in Figure 6.

The effect of water concentration

An air velocity of 200 ft/s with the fuselage set for level flight was employed to investigate the effect of water concentration at air temperatures of $-5^{o}C$ and $-10^{o}C$.

At $-5^{o}C$ the water concentration was varied between 0.15 g/m^3 and 0.7 g/m^3 and Figure 7 shows the ice deposits accreted after 30 minutes icing at three water concentrations. The "A standard" mushroom front face (left of picture) stayed clear of ice at all conditions, but there were ice deposits on the intake flare at all conditions with more on the bottom than on the top. The "B standard" mushroom had ice on its front face at all the water concentrations tested with the amount increasing as the water concentration was increased.

At $-10^{o}C$ the water concentration was varied between 0.15 g/m^3 and 0.6 g/m^3 with results similar to those at $-5^{o}C$ except that the ice deposits were slightly heavier.

The effect of air temperature and aircraft attitude

The two effects of air temperature and aircraft attitude were most conveniently investigated together at water concentrations set to the specification for maximum continuous operation.

Dealing first with the level flight aircraft attitude, the air temperature was varied between -2°C and -15°C and results for three air temperatures are shown in Figure 8. In the case of the "B standard" intake the mushroom front face remained clear of ice down to a temperature of -3°C, but at -4°C considerable deposits were produced. The intake flare remained clear of ice down to -2°C but as the air temperature was further reduced the ice built up at a progressive rate. The "A standard" mushroom front face stayed clear of ice down to -10°C, and the intake flare had ice accumulations similar to the "B standard" intake.

Figure 9 shows representative results obtained from the 10° nose up attitude tests which were carried out at air temperatures between -2°C and -10°C. The "B standard" mushroom front face stayed clear of ice at -3°C and at -4°C only a few small pieces were observed on the top half. At lower temperatures the ice build-up was quite considerable. The intake flare remained clear down to -3°C, but increasing amounts of ice were produced on the lower part of the intake at the lower temperatures. It was noted that the ice deposits on the lower half of the flares and forward engine cowling were greater than in the level flight case. The "A standard" mushroom remained clear of ice on the front face at all conditions down to -10°C, but the intake flares accumulated ice below -3°C in a similar fashion to the "B standard" mushroom.

With the fuselage in the 10° nose down attitude, tests were again carried out at air temperatures between -2°C and -10°C and representative results are shown in Figure 10. The "A standard" mushroom front face remained clear of ice at all test conditions, but the intake flare had ice deposits at -4°C on its lower half. As the temperature decreased down to -10°C so the amount of ice increased, and it was considered that the amount was greater than in the corresponding level flight tests. The "B standard" mushroom remained clear of ice during the -3°C test, but a moderate accumulation occurred at -4°C and this increased as the temperature was lowered. The intake flare behaved in a very similar manner to the "A standard" intake.

Ice ingestion

On a number of occasions during icing and melt-off sequence, ice ingestion into an intake was observed. The trajectories were

analysed from video tape as follows: (for each incident the air velocity, air temperature, and water concentration during ice deposition are given).

During icing, a piece of ice became detached from the port mushroom front face and entered the starboard intake; 200 ft/s, $-5^{o}C$, 0.7 g/m^3.

During melt-off, ice on the front face of the port mushroom became detached and broke into two pieces. The top piece lifted clear over the top of the intake, but the bottom piece fell on to the cabin roof behind the snow fence and broke into a number of pieces. These pieces remained moving about in the turbulent zone behind the snow fence for some time and one piece lifted up into the port intake; 200 ft/s, $-5^{o}C$, 0.7 g/m^3. This sequence is shown in Figure 11.

During melt-off, a large piece of ice on the inboard section of the port intake flare became detached and entered the port intake, 200 ft/s, $-5^{o}C$, 0.7 g/m^3. This sequence is shown in Figure 12.

During melt-off, ice on the front face of the port mushroom became detached, fell downwards and some pieces entered the port intake; 200 ft/s, $-15^{o}C$, 0.45 g/m^3. This sequence is also shown in Figure 12.

The two danger areas where an accumulation of ice could lead to ice ingestion are: (a) the mushroom front face, (b) the intake flare.

Ice begins to form on the unheated mushroom front face of the "B standard" intake and also on the intake flare at air temperatures below $-2^{o}C$. Operation of this particular intake would therefore appear to be unsafe in icing conditions below this temperature. While the mushroom heated front face of the "A standard" intake remained clear of ice down to $-10^{o}C$, ice started to accumulate on the intake flare at temperatures below $-2^{o}C$. Operation of this intake would also appear to be unsafe in icing conditions below $-2^{o}C$.

All the melt-off sequences were made with an air velocity of 200 ft/s at which condition most of the ice shed from the front of the aircraft fuselage passed over and outboard of the engine intakes. At lower airspeeds the chance of ice striking an intake will be increased but the mushroom should prevent it from being ingested.

Tests with a Gnome H1400 engine installed in the port nacelle

Some icing tests were made with a Gnome engine installed in the port nacelle with ejector simulation on the starboard side. Two "A standard" mushrooms were fitted and these were modified by disconnecting the electrical anti-icing on the mushroom front faces to approximate to "B standard" intakes. The engine had a locked power turbine and was run at a shaft speed of 25,564 rev/min and an engine air mass flow of 13.2 lb/s.

The tests were made at an air velocity of 200 ft/s in level flight, and Figure 13 shows that at an air temperature of -5°C and water concentration of 0.7 g/m^3 there was less ice deposited on the forward facing lower nacelle panels than on a similar test with two ejectors. It is thought that this is a result of heat radiation forward from the engine. At an air temperature of -10°C this effect was not detectable.

Water droplet distribution tests

To investigate the water droplet distribution at the blowing nozzle exit, a coarse pitch wire screen was clamped to the exit flange so that the droplets from the spray rakes were deposited on the wire to give an indication of distribution. Figure 14 illustrates the distribution at an air velocity of 200 ft/s, air temperature of -5°C and a water concentration of 0.7 g/m^3. It can be seen that an acceptably even distribution was achieved at test conditions representative of those of the majority of the tests.

During the trials it was noted that higher ice accumulations were present below and outboard of the engine intakes but elsewhere the distribution was considered to be even. To check this distribution a screen similar to the one used at the blowing nozzle exit was rigged over the helicopter cabin roof in front of the engine intakes. A test was made at the same conditions as those with a screen at the blowing nozzle exit, and the result, also illustrated in Figure 14, confirmed the pattern of locally increased water concentration due to the presence of the helicopter fuselage. It should be borne in mind that some of the deposited ice fell off the screen when the airflow was being shut-down prior to taking the photographs.

Snow fence

For the first eight tests the aircraft was fitted with a curved snow fence fixed to the top of the fuselage in front of the engine intakes. This was to prevent snow accumulation on the cabin roof from slipping backwards and being ingested by the intakes.

Tests showed however, that any pieces of ice getting behind the snow fence remained in a re-circulation zone close to the intakes for a long time, thus increasing the chance of ingestion. In view of this the snow fence was removed.

Conclusions made from Sea King tests

The tests indicated that air temperature was the parameter having the largest influence on ice deposition; a temperature drop of only 1 or 2°C was found to result in a change from no ice deposition at all to very heavy ice accretion requiring, in flight, immediate exit from these conditions. Ice accretion is of course dependant on water concentration and entry into Intermittent Maximum can be a potential hazard in a very short time. Whilst a change in aircraft attitude will result in local changes in ice distribution, it will have little overall effect and therefore has the smallest influence on ice deposition.

Details of Lynx helicopter icing trials

Whilst the Sea King was used for a general investigation into helicopter icing problems, tests on the Lynx fuselage were an initial survey aimed at determining modifications required for a flight icing clearance.

General views of the Lynx helicopter installation are shown in Figure 15. Two ejectors were again used to induce airflow through the engine intakes and this could be varied to provide engine SHP simulation between flight idle and single engine contingency. Two smaller ejectors were fitted to simulate the airflow through the oil coolers, and hot air was piped into the rotor gearbox cowling and fuselage to simulate oil heating and cabin heating respectively.

The engine intake anti-icing consisted of electric heater mats having a total power rating of 3.8 kW for each intake.

All water concentrations were for the maximum continuous rating, with 30 min test duration.

The effect of air temperature change

The effect of air temperature change was investigated with the fuselage positioned 5° nose down, air velocity 203 ft/s and simulated 400 SHP.

Except for the continual build-up and shedding of ice from the intake leading edge joints, the inner surfaces of the intakes

were clear of ice at air temperatures of -2°C and -5°C, but not clear at -8°C and -15°C. The ice deposits seen inside the intakes at -15°C were more widely spread and significantly heavier than at the higher temperature.

All the tests demonstrated that ice readily accumulated on the leading edges of the oil coolers and gearbox cowling (Figure 16), with the heaviest ice accretion on the oil coolers occurring at -5°C. Ice deposits on the gearbox leading edges increased as the air temperature decreased and at -15°C the ice was of a white, spiky, brittle nature.

At -2°C the windscreen top ledge was generally clear of ice, but there were small nodules on the windscreen central dividing strip, and hard clear ice on both windscreen wiper arms. With decreasing temperature down to -15°C, the accumulation on the top ledge and wiper arms became progressively heavier, but at no time was the windscreen completely covered with ice.

The effect of forward velocity

Increasing the forward velocity to 270 ft/s with the fuselage 5° nose down and air temperatures of -8°C and -15°C produced very heavy ice accumulations on the engine intakes, and as it was seen to be unrealistic for the aircraft to operate at these conditions no further tests were made at this velocity during this initial test series.

Ice ingestion

During the icing tests and subsequent melt-off sequences, ice was seen to detach from the aircraft structure and either enter an intake or pass close by.

Figure 17 shows a detachment during icing, from the starboard side of the gearbox top lip, enter the top part of the starboard intake, 203 ft/s, -5°C, 0.7 g/m³.

During melt-off ice shed from the lower leading edge of the gearbox cowling and entered the starboard intake, 236 ft/s, -5°C, 0.7 g/m³. This sequence is shown in Figure 18.

In the case of the Lynx, potential hazards from engine ice ingestion were produced by the rotor gearbox cowling and the unsmoothed windscreen top ledge. Removing the bluff areas on the gearbox cowling and smoothing the windscreen "eyebrows" gave a marked reduction in ice accretion, and hence reduced the possibility of engine ice ingestion.

Further changes in these areas have since taken place and repeat trials in the facility are planned to take place shortly.

Concluding remarks

A brief description has been given of a test cell at NGTE in which icing trials may be undertaken on full-scale helicopter fuselages. Some results have been presented from extensive icing trials made on the front portion of Sea King and Lynx helicopters, over a range of air temperature, water concentration, aircraft attitude and forward velocity. The anti-icing performance of the engine air intakes was explored and the ice ingestion characteristics noted during icing and melt-off sequences.

The authors wish to express their thanks to colleagues at NGTE who participated in these trials and in particular Messrs. R. G. J. Ball and D. C. Dransfield.

Further we wish to acknowledge the valuable advice and co-operation freely given by members of Staffs of Messrs. Rolls-Royce Ltd., Westland Helicopters Ltd., and the Aeroplane and Armament Experimental Establishment.

Finally our thanks are due to the Procurement Executive, Ministry of Defence for permission to publish this Paper. The view expressed are entirely our own.

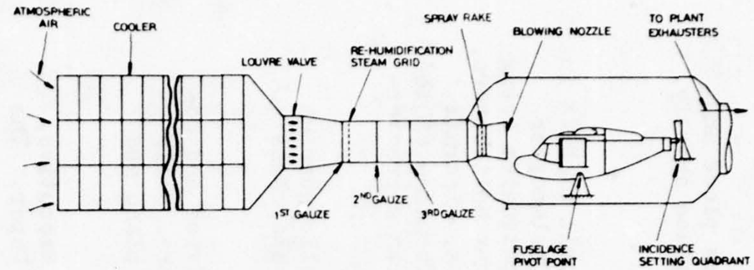


FIG. 1 ARRANGEMENT OF LYNX HELICOPTER FUSELAGE IN CELL 3 WEST

FIGURE 1 Arrangement of Lynx Helicopter fuselage in Cell 3 West

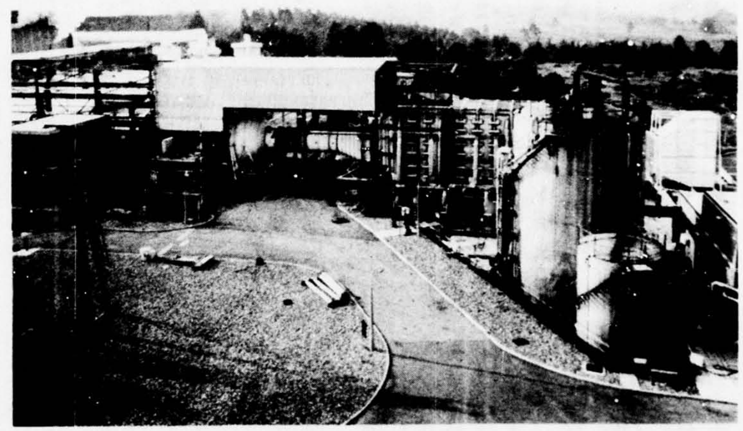


FIGURE 2 General view of Cell 3 West

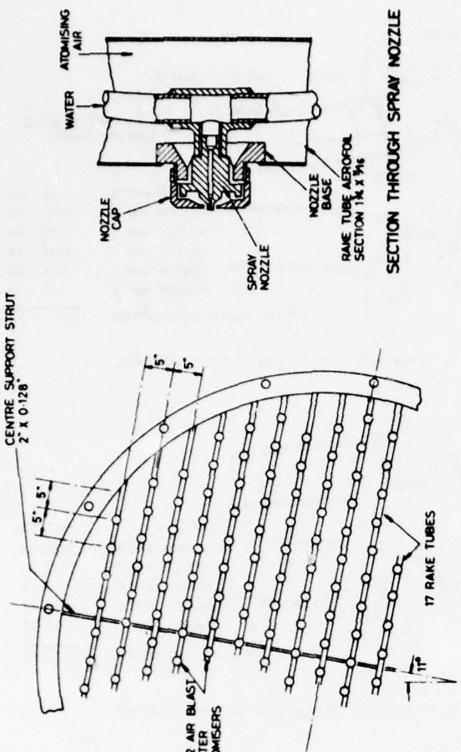


FIG.3 ARRANGEMENT OF WATER SPRAY RAKE

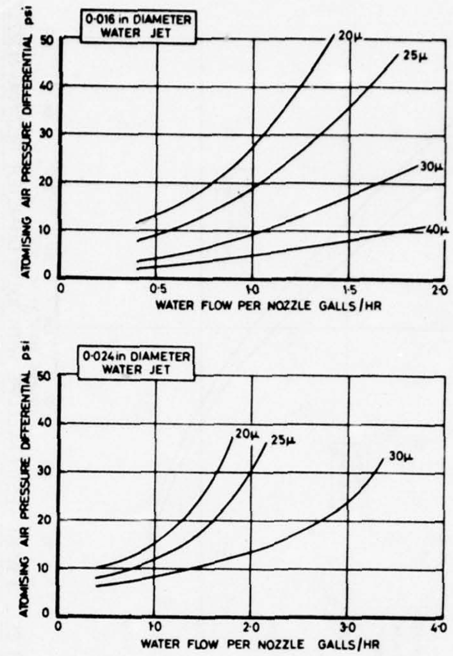


FIG.4 CHARACTERISTICS OF SPRAY RAKE AIR BLAST ATOMISERS

FIGURE 3 Arrangement of water spray rake

FIGURE 4
Characteristics of spray rake air blast atomisers

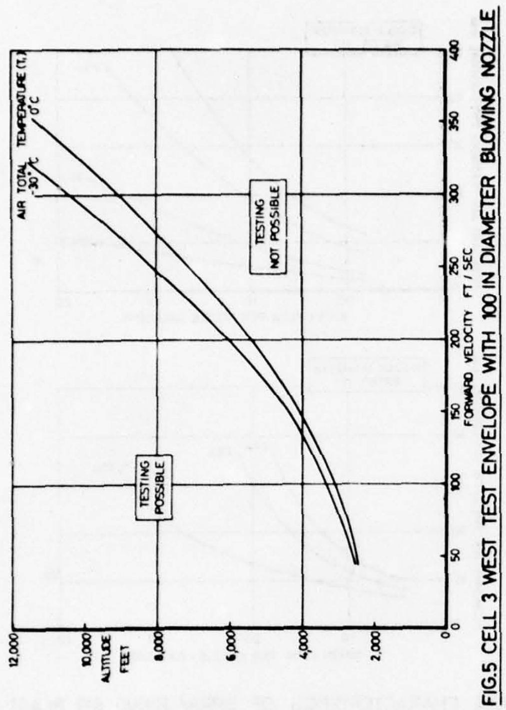


FIGURE 5 Cell 3 West test envelope with 100 in. diameter blowing nozzle

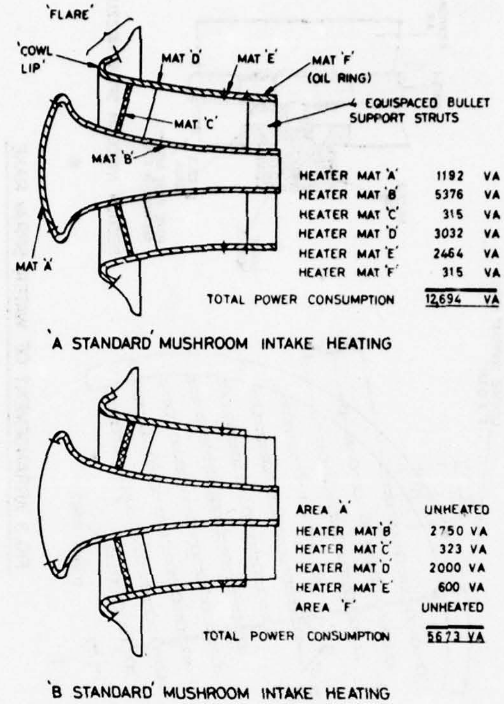
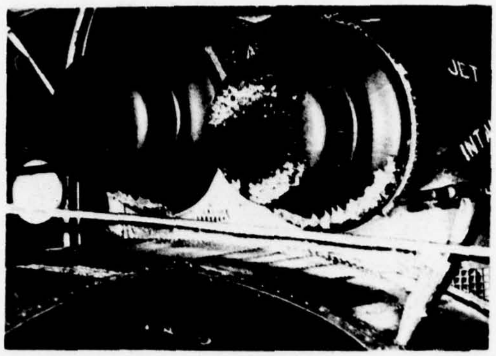
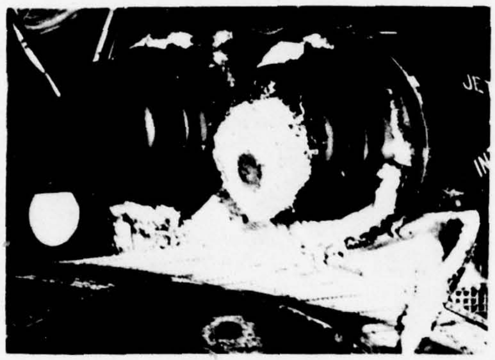


FIG. 6 MUSHROOM INTAKE HEATING ARRANGEMENTS

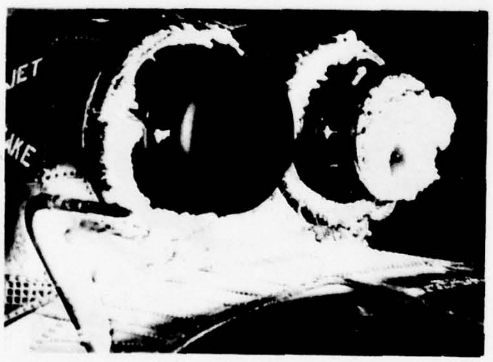
FIGURE 6 Mushroom intake heating arrangements



0.15 g/m^3



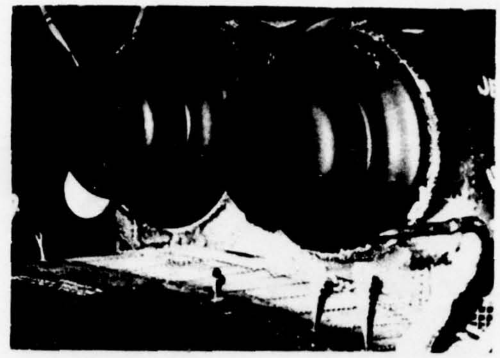
0.45 g/m^3



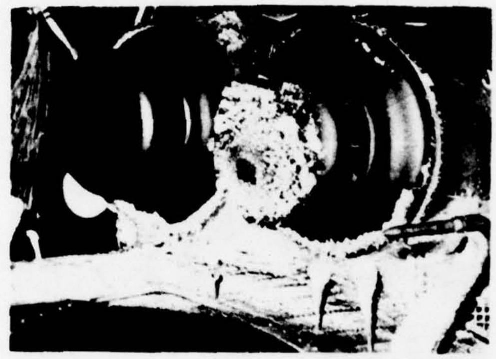
0.7 g/m^3

Level flight
Forward velocity 200 ft/s
Air temperature -5°C

FIGURE 7 The effect of water concentration on ice deposition



-3°C

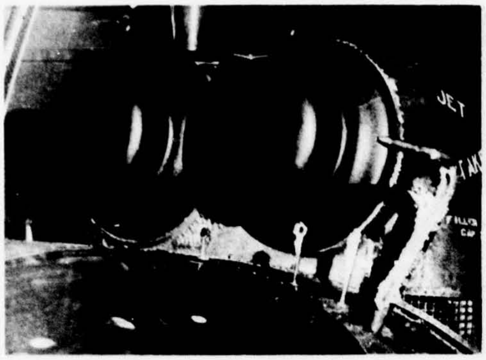


-4°C

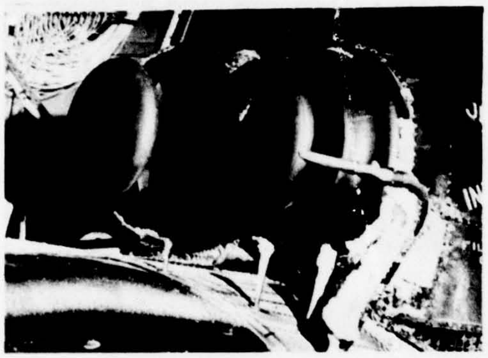
-5°C

Level flight
Forward velocity 200 ft/s
Continuous maximum water concentration

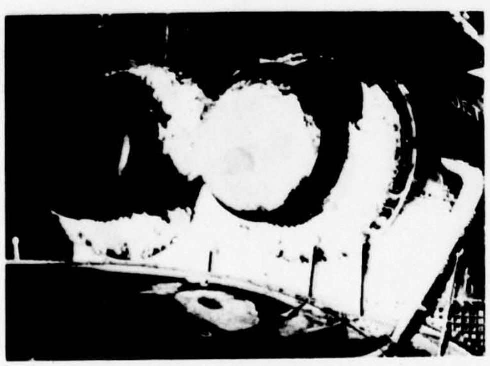
FIGURE 8 The effect of temperature on ice deposition



$-2^{\circ}C$



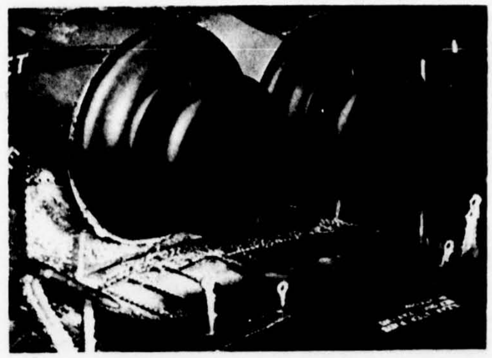
$-4^{\circ}C$



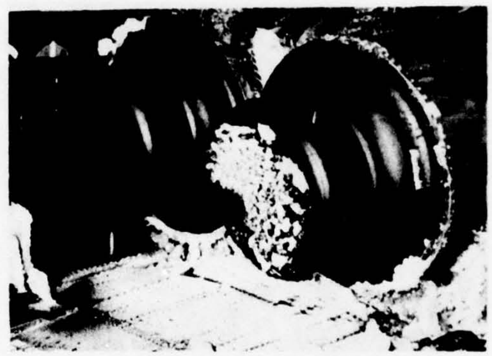
$-10^{\circ}C$

10° nose up
Forward velocity 200 ft/s
Continuous maximum water concentration

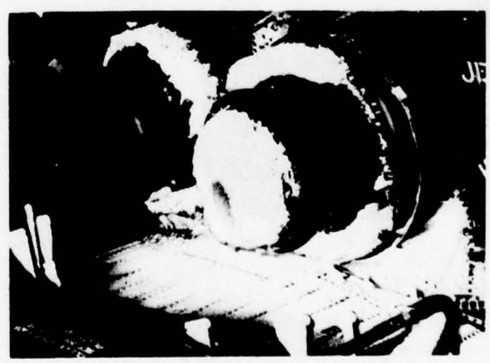
FIGURE 9 The effect of temperature on ice deposition



$-2^{\circ}C$



$-4^{\circ}C$



$-10^{\circ}C$

10° nose down
Forward velocity 200 ft/s
Continuous maximum water concentration

FIGURE 10. The effect of temperature on ice deposition

Start of events

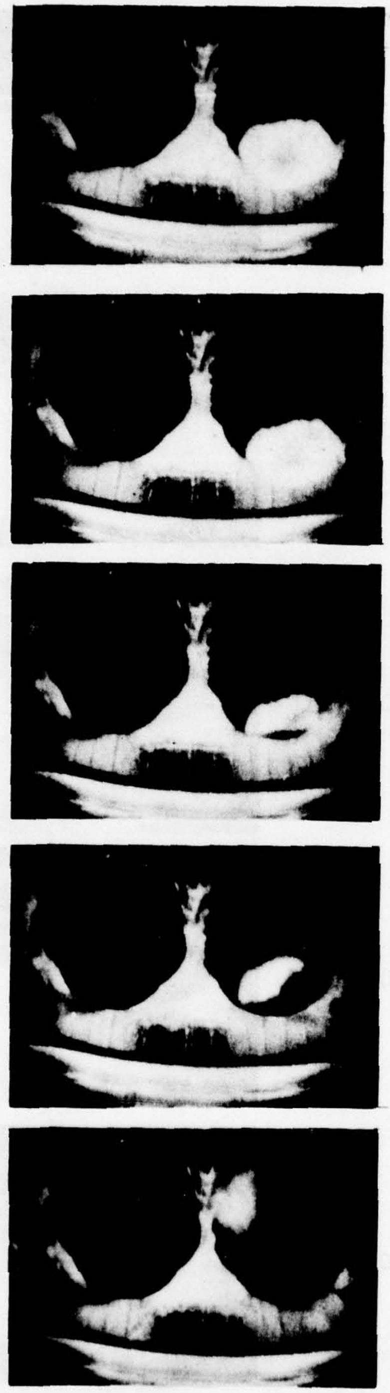
0 s

+0.24 s

+0.26 s

+0.32 s

+0.38 s



Ice ingestion from front face of intake after 30 m at $-5^{\circ}C$ and 0.7 g/m^3

Start of events

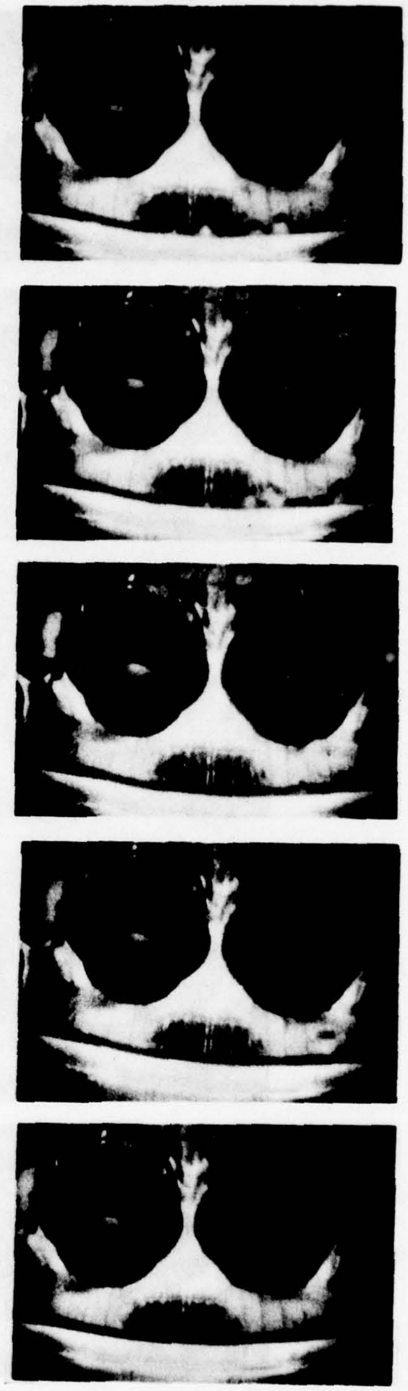
0 s

+0.14 s

+0.22 s

+0.38 s

+0.48 s



Retention of ice particles behind snow fence after shedding from front face of intake

FIGURE 11 High speed television sequences

Start of events

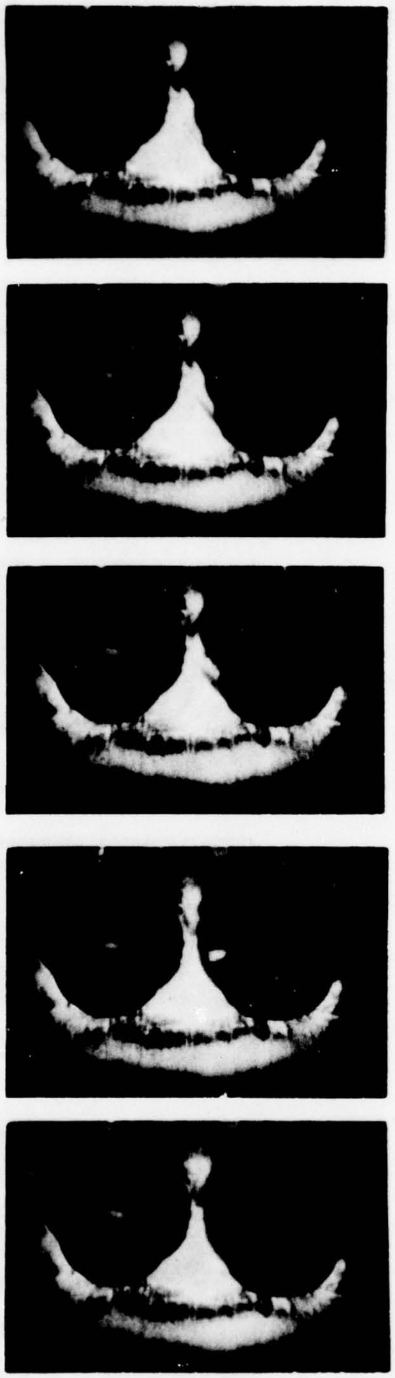
0 s

+0.04 s

+0.06 s

+0.10 s

+0.12 s



Ice ingestion from lip of port intake after 30 m at -5°C and 0.7 g/m^3

Start of events

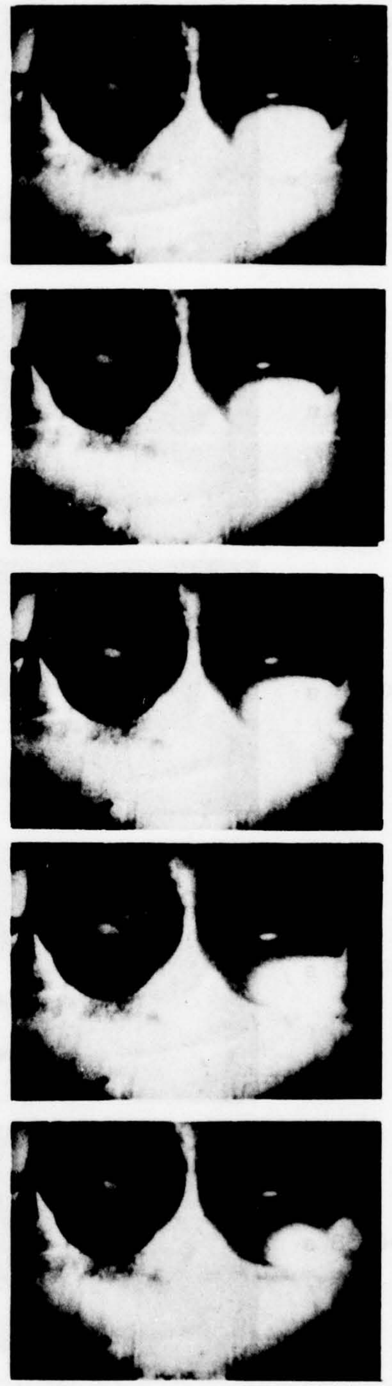
0 s

+0.02 s

+0.06 s

+0.10 s

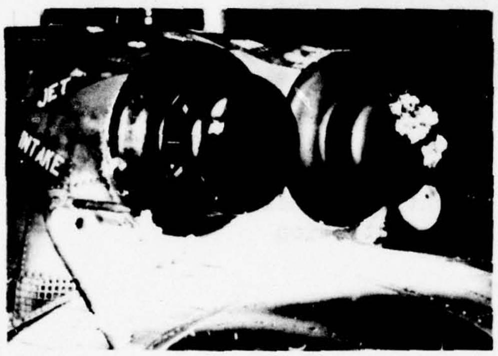
+0.12 s



Ice ingestion from front face of intake after 30 m at -15°C and 0.45 g/m^3

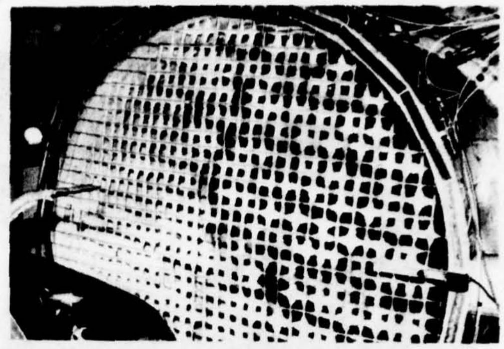
FIGURE 12 High speed television sequences

Starboard nacelle - ejector simulation
Port nacelle - Gnome H1400 engine
15 m at 0.7 g/m^3, $-5^{o}C$,
200 ft/s.

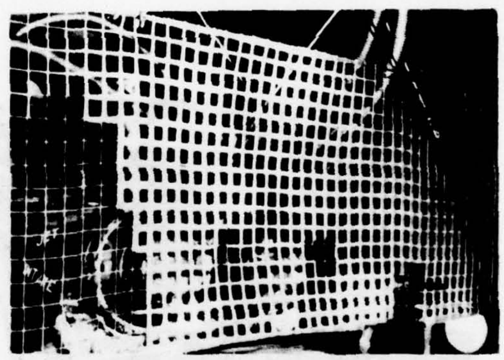


Starboard nacelle - ejector simulation
Port nacelle - ejector simulation
15 m at 0.7 g/m^3, $-5^{o}C$,
200 ft/s.

FIGURE 13 The effect of heat transfer from Gnome H1400 engine to intake fairing.

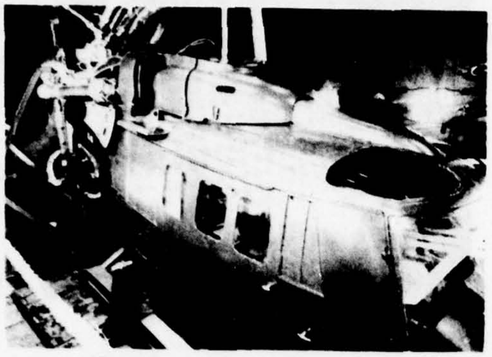


Demonstration of ice deposit distribution at nozzle exit plane.

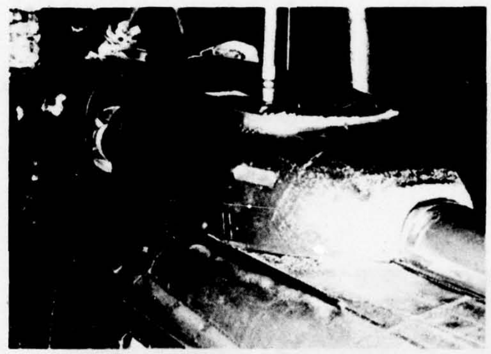


Demonstration of ice deposit distribution in front of engine intakes.

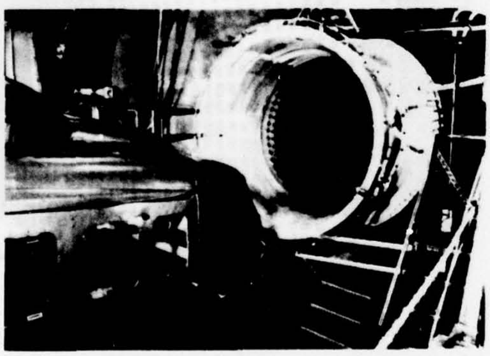
FIGURE 14 Ice distribution tests.



View looking downstream



5^{o} nose down
Forward velocity 203 ft/s
Air temperature $-5^{o}C$
Continuous maximum water concentration



View looking upstream

FIGURE 15 Installation of Lynx in Cell 3 West.

FIGURE 16 Ice accretion on gearbox sliding cowling leading edges.

Start of events

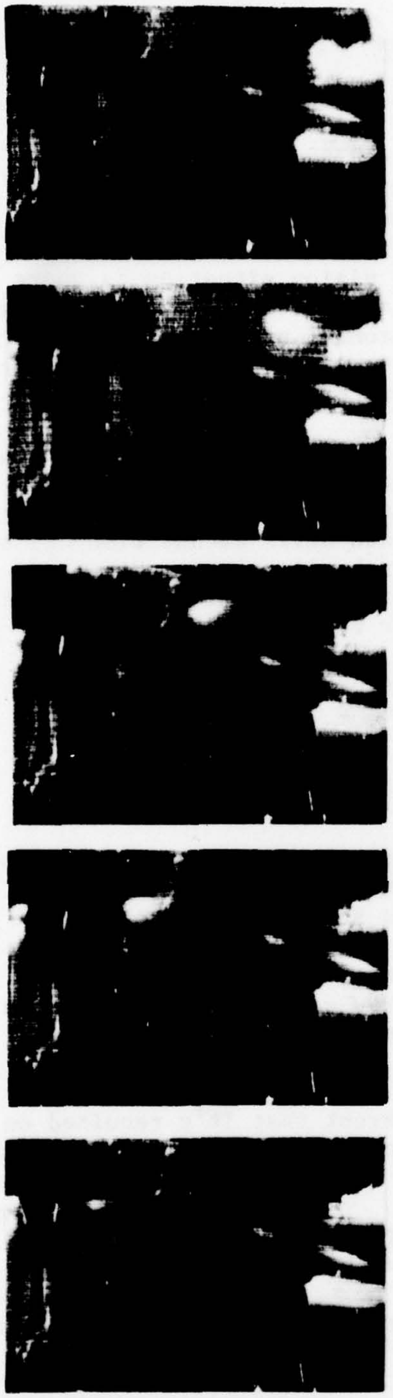
0 s

+0.04 s

+0.09 s

+0.13 s

+0.15 s



Ice ingestion from starboard side of the gearbox cowling lip after 22 m of icing at $-5^{\circ}C$ and $0.7\ g/m^3$

FIGURE 17 High speed television sequences.

Start of events

0 s

+0.08 s

+0.10 s

+0.12 s

+0.13 s

Ice ingestion from starboard side of the lower leading edge of gearbox cowling after 14 m 50 s of melt off at $-5^{\circ}C$ and $0.7\ g/m^3$

FIGURE 18 High speed television sequences.

DISCUSSION

6 June 1974 A.M. Session

Dr. Rosen: The situation forces difficult decisions. Many of us are sitting here and asking ourselves, well what does all this mean? What I would like you to do, if possible, is to go down these things one at a time. Forward vision situation is quite clear. I don't think that requires any discussion. Let's talk for a moment about the tail and horizontal stabilizer. In your opinion do you feel that these devices, that these components require either anti-icing or deicing protection?

Mr. Wilson: Our experience to date is that neither the horizontal stabilizer nor the tail rotor requires any protection. Although on many flights we've accrued large quantities of ice on the leading edge of the stabilizer, we have never at any time had any handling problems which could be attributed to this. As I mentioned earlier the tail rotor invariably gives clean shedding. We have never detected any tail rotor vibration in flight.

Dr. Rosen: I surmised from your presentation that you are not ready to say that thermoelectric systems are the panacea for rotor design?

Mr. Wilson: That is correct.

Mr. Plackis, FAA: In this discussion and also in our discussions Tuesday I think it was rather apparent that the need for tail rotor protection was not indicated as strongly as main rotor. However, the film shown yesterday in the presentation on HH-53 indicated more than just a casual need for protection in that area. Is there any apparent reason you can see why it would be more desirable to protect that aircraft than the ones you've experienced?

Mr. Wilson: Why do you say it's apparent that it's required on the 53? Surely that's been given a clearance for flight without any rotor protection?

Mr. Plackis: I believe that the gentleman that came from El Centro showed considerable buildup on the tail rotor?

Mr. Wilson: Yes, but there was never any suggestion that he had any control problems, at least if he did I didn't pick it up. What he did say, of course, was they saw a lot of tail rotor damage from ice shed from the main rotor, but that's a different matter.

Mr. Plackis: I was under the impression that there was a considerable buildup on the tail rotor also, perhaps not.

Mr. Wilson: No, that message didn't come across to me anyhow. Thank you.

LT Jaeger, AFFTC: I think that what Major Loviern said was that we found about a 1/4" buildup on the tail rotor before it would shed and that they tried the ADP tape on it and the tape they tried wasn't suitable for the aircraft and that's when they had appreciable buildup on the tail rotor. We have no intention of requesting or recommending additional tail rotor deicing. As far as ADP is concerned, I think Sikorsky themselves will agree that it's best at flush. Am I right Ken? - answer: Yes. There you have another opinion.

CPT Checketts, RN: I think we'll have to make this the last question. To amplify your answer to Mr. Rosen about main rotor heating. In spite of the doubts which exist about its efficiency at the moment, and the faults which we've discovered, and deficiencies which have become apparent here, I'd like to assure Mr. Rosen that we are continuing to, we hope if we can get the money, to develop the system; and we hope at least to get one more season's work out of it. We haven't rejected the concept, we are going ahead, we hope to develop what will be a satisfactory system.

Mr. Wilson: Yes, I certainly didn't intend to give you that impression. I'm sorry if I did. I'm talking about the present stage of development of heated rotor blades, the ones I've seen, that's all.

DISCUSSION

CPT Checketts: It didn't come over quite clearly to me to what extent you could test a helicopter in a free air steam with snow. Do you have that capability? In snow?

Mr. Tolliver: In snow we can, as far as snow, we can do direct snow, if we're doing direct snow injection as far as the engine qualification, we like to just blow it into the engine. (Duct it in?) Right. We can supply snow clouds and blow snow at the entire aircraft, if that's what you want, but normally we try to control it to just the engine. (Up to what speed?) Up to presently about 150 knots.

CPT Checketts: Are you looking for work?

Mr. Friedlander, French A.F.: How do you produce snow or ice crystals?

Mr. Tolliver: We have two basic methods to produce snow. We use, what's an ice crystal really. We use atomized water in different sizes, some of them quite large. We also have several methods to make snow without using compressed air and employing turbine blades. They create a low pressure area in front of the blade and blow water through it and so on. Those are the 2 basic methods. I'd be glad to talk to you more about it later if you wish.

DISCUSSION

Dr. Rosen: I would like to compliment our guests from West Germany for what seems to be an outstanding design. I'd like to ask some basic questions. Yesterday, Mr. Schmidt from Lockheed indicated that perhaps the best way to design a system was with a cordwise variation and you seem to have spanwise elements very similar to our CH-3 design; and I was wondering if you might offer some comments as to one design philosophy compared to the other?

Mr. Horlebein, M.B.B.: We have made some investigations about that problem and based on the experience of the N.R.C. we came to the decision that although there is perhaps a little bit more power necessary, it has production advantages to take the system.

Dr. Rosen: Yes, I quite agree with you, thank you.

Mr. Lewis, AEFA: I wonder if you would comment as to whether you think the soft in-plane design assists you in shedding ice at the relatively warmer temperatures? One thing that we have noticed, I think on the Hueys, is that with their very stiff blade they will accumulate ice at above -5°C; whereas you have indicated in all of your tests that there is little degradation due to ice accumulation above -5°.

Mr. Horlebein, M.B.B.: If I heard your question right, you can say we can stand temperature minimums in dry air and icing down to -5° and we have no ice accretions. We feel the rotor tip speed contributes to lack of ice accretion.

Mr. Lewis: So you are attributing the ability to the adiabatic temperature rise on the blade as opposed to any flexibility which you buy by your soft in-plane design?

Mr. Wilson: In view of what we've heard today Boscombe Down has seen ice for considerable distances toward the tips, at temperatures of -2 or -3°, and we have had 1 or 2 fairly exciting occurences at these temperatures. Do you feel maybe that the simulation wasn't correct and what ideas have you got in order to avoid run back if you find you have to do it?

Mr. Horlebein, M.B.B.: You mean temperatures deeper than -5°? (Yes.) We didn't encounter any problems during that test and we have no camera on the rotor head to have a look at what happens there on the blade; but we didn't encounter mentionable power increase or any loss in control power; but we are sure we must make additional tests; but up to now we have a good confidence to do our job in that range. That's what I can state.

Jerry Werner, Lockheed: I have 2 questions for you. The first is,

it wasn't clear to me, are you using DC on the heater elements or AC?

Mr. Horlebein, M.B.B.: We used DC on the heater elements. AC is converted to DC in the aircraft.

Mr. Werner: What is the efficiency of your conversion system from AC to DC? In other words, what is your generator power compared to; what do you get out on the blades?

Mr. Horlebein, M.B.B.: It is very high. I don't know the exact number at the moment but they are near comparable.

Mr. Werner: My other question, can you describe the design of the heating element. I'm interested in how much insulation you have between the erosion shield and the heating elements?

Mr. Horlebein, M.B.B.: The insulation layer of plastic on the blade is about 1/10 millimeter thick and it changes because we have a graded system.

Mr. Werner: That's the heating element itself? How much insulation do you have between the heating element and the outer erosion shields?

Mr. Horlebein, M.B.B.: On the erosion shield, its about 0.5 millimeters, including adhesive.

DISCUSSION

Alex Petach, Hughes Helicopters: Mr. Swift, you made a comment, I wonder if you could elaborate; since you passed over it and this was, with regard to examining engine case heat transfer. What was it that you were actually trying to look at? Could you just elaborate on that phrase?

Mr. Swift: What we were looking for to see if there was any effect on the engine due to helping anti-ice the region around the intakes.

Mr. Petach: I see, in other words, merely the heat transfer (engine without specific anti-icing). Very good. Now I don't have a question, but I have a comment for Mr. Horlebein and he prepared an excellent paper and I want to thank him for it; but by the time it gets printed by the Army, I wonder if I could make one more request and that is a cut-away or a slide with a few words directed towards your engine installation. You passed over that very quickly; and although it worked very well I think you might get some attention from people if you added that. Thank you.

(The following discussion was too far from the microphone to transcribe.) Thank you very much.

CLOSING REMARKS

COLONEL DEAN E. WRIGHT
Commander, USAAEFA

The first thing, I feel we have achieved the goals of the conference in the dissemination of information. I know this will go on the rest of the afternoon and probably a couple of days before we get done, but I think we've done a wonderful job there. The questioning has been extensive, we've had a number of informal discussions and as a result of this although there is a lot yet to be done. A lot of effort directed toward the solution of the icing problems and the results support each other. In particular, we have, of course our own particular requirements which have to be satisfied. If I can summarize those requirements, we need rotor system protection, we need windscreen protection, we need engine inlet protection, and we need some type of icing severity indicator so that in the final analysis the pilot will know where he is in the icing regime. During the past 2 1/2 days, we have had a number of solutions prepared, some have worked, others have not. I think that we'll continue with that effort. The one thing I feel that we need as a developer is some input from the user as to what he will accept. Will he need his full windscreen clear of ice or will he accept a lesser? What does he need in the deicing, what does he need in the engine protection? Yes, we can go exotic and give him all the protection we feel as engineers are needed to fully protect the system, but I think the user has to come up with his requirements and tell us just how far he wants us to go. The one thing, and I'm glad our friends from Germany brought it out that in order to operate under all weather conditions, you need an aircraft that is IFR certified. We need an IFR capability in addition to an anti-icing/de-icing capability. We are directing our efforts in that direction, and I know the other services that are represented here are doing that also.

In summary, we do need to improve our interchange. We need to keep it going on a continuing basis. We need to know what the other services, what the other countries, are doing in their testing and we also have to know what the equipment developers, what the manufacturers are planning. If we can keep the interchange going, I think this was the start of it, we will achieve our final results. In that vein, I invite each of you to attend our tests, witness our testing which will commence in October. If you have specific requirements that you would like to have included in our testing, either contact the Activity or contact AVSCOM and if we can accomodate your requests in our tests we would be more than glad to and for the manufacturers, if you have equipment or products that you would like us to look at, please let us know.

I would like to thank each of the session chairmen, and all the authors for their excellent papers. I think that we've had, every one of our papers has been excellent in its field. Before continuing, I would like to ask each of the session chairman if they have anything else.

Colonel Beasley, do you have any comments you would like to make?

Colonel Beasley: I'd just like to reinforce what you've said about the quality of the papers and presentations. I think they have been outstanding in every respect. Personally, I think I've gained an awfully lot in this symposium, I really didn't understand when I came out here and I confess this, - the scope and depth that I got from the presentations. I will say this about our directions and what we are trying to do. We can indeed do a whole lot with technology, but I think we all know that, but you always have to bear in mind impact, complexity, cost, what it does to the Commander; the guy that has to operate these systems in the field. He needs the capability to operate in all weather. Don't put an anvil around his neck in trying to give it to him. That's the cross we bear when we are developing a system.

Captain Checketts, do you have any comments you would like to make?

Captain Checketts: Perhaps the only thing I would say is, we've seen some differences of opinion on the need or perhaps the level of rotor blade protection which is required amongst the different participants. One thing perhaps that we haven't touched at all is what scale and scope of tests is needed to certify a rotor blade protection system. The tests that seemed to have been done in the past perhaps elsewhere than in England have only explored perhaps the envelope in a very limited way and maybe here is the ground for future discussion between the experts in the various countries to determine a standard way in which these tests could be carried out in the future.

Mr. Wilson: I really have not very much to say other than to thank you very much for inviting me here and permitting me to present the paper. I've learned quite a lot from all the other papers, and I've sure found it most instructive and enjoyable. I support what my two previous speakers have said, I think we certainly do require to get together on establishing our common aim requirements. What we should be trying to meet and on matters of testing and so forth. This, I think is quite a lot of you know there is already some action going ahead I hope in this direction.

Dr. Haneman, do you have any comments you would like to make?

Dr. Haneman: The work that Colonel Wright and his people have done here in putting this program together for us and the facilities and

the operation, I think are just as commendable, if not more so, than all the fabulous work that has been done. So, I would like to take this opportunity to personally thank you and the troops for a very well done job.

Thank you very much. I can only do what my people do for me. I had good people on this. First, to the Flight Test Center and especially to the Officer's Club thank you for the support given us. Everything we have asked for they have come through willingly, eagerly and given us. A special vote of thanks goes to Jimmy Hayden, Ed Travaras, Linda Gunderson, Hank Hurtt, Bob Buchanin, and of course the other girls at the desks. They worked a long time on this and just to make sure it would go smoothly. We will be assembling the minutes for printing in early July, so I ask those authors who are taking their papers back with them, if they would get them in as soon as they could so we can get them printed.

Any additional comments, questions from the floor?

There will be a tour of the flight line leaving the Club at 1315 hours, and would be about one hour returning to the Club. Also, any of you who do not desire to go on the flight line tour or have additional time, I invite you down to the Activity, Building 1820.

I want to thank all of you for your attendance, your interest, your searching questions and comments, and I do hope that sessions of this type can be repeated.

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Director, US Army Air Mobility Laboratory
(Acting Director of RD&E, AVSCOM)

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Chief, Flight Standards Division, AVSCOM

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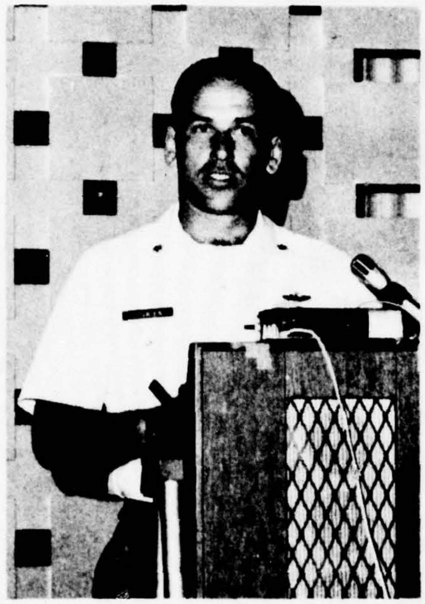

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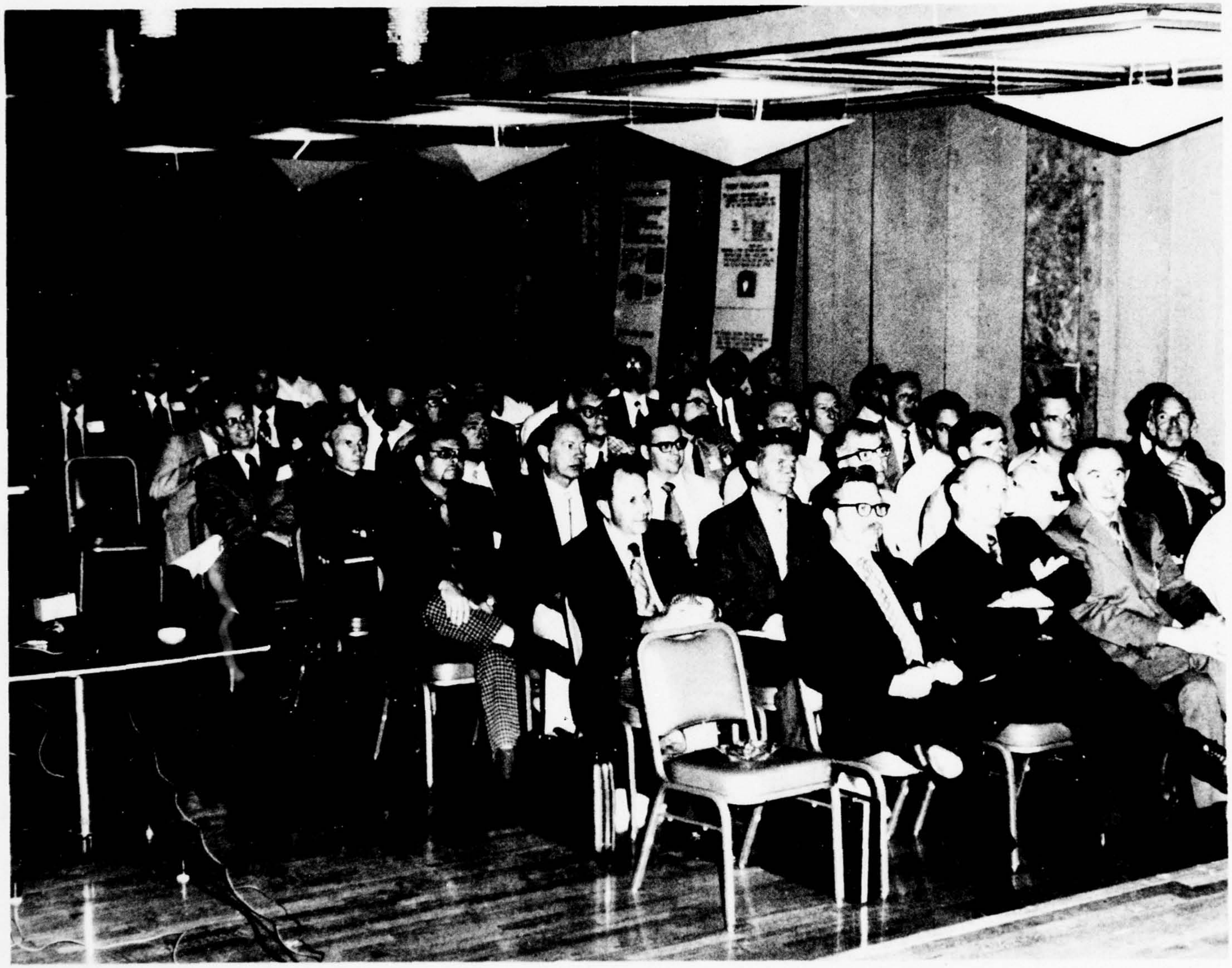

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